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NASA/LOCKHEED
AIRCRAFT ENERGY EFFICIENCY

ACEE COMPOSITE STRUCTURES TECHNOLOGY

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Papers by
Lockheed-California Company
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Burbank, California 91520

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FOREWORD

The NASA Aircraft Energy Efficiency (ACEE) Composite Primary Aircraft Structures Program has made significant progress in the development of technology for advanced composites in commercial aircraft. Under NASA sponsorship, commercial airframe manufacturers have now demonstrated technology readiness and cost effectiveness of advanced composites for secondary and medium primary components and have initiated a concerted program to develop the data base required for efficient application to safety-of-flight wing and fuselage structure. Timely dissemination of technical information acquired in these programs is achieved through distribution of reports and periodic special oral reviews.

The third special oral review of the ACEE Composites Programs was held in Seattle, Washington, on August 13-16, 1984. The conference included comprehensive reviews of all composites technology development programs by ACEE Composites contractors - Boeing, Douglas, and Lockheed. In addition, special sessions included selected papers on NASA-sponsored research in composite materials and structures and reviews of several important Department of Defense programs in composites.

Individual authors prepared their narrative and figures in a form that could be directly reproduced. The material is essentially the same material that was orally presented at the conference. The papers were compiled in five documents. Papers prepared by personnel from Boeing Commercial Airplane Company, Douglas Aircraft Company, and both Lockheed-California Company and Lockheed-Georgia Company are contained in NASA CR-172358, CR-172359, and CR-172360, respectively. Papers on selected NASA-sponsored research are contained in NASA CP-2321. Papers on selected Department of Defense programs in NASA CP-2322.

The assistance of all authors, contractor personnel, and the Research Information and Applications Division of the Langley Research Center in publishing these proceedings is gratefully acknowledged.

John G. Davis, Jr.
Technical Chairman for
ACEE Composite Structures
Technology Conference
Langley Research Center

CONTENTS

ACEE COMPOSITE STRUCTURES TECHNOLOGY CONFERENCE

AGENDA

MONDAY, AUGUST 13, 1984

[1]SESSION 1: OUTLOOK FOR COMPOSITES IN FUTURE AIRCRAFT

SESSION CHAIRMAN: Robert L. James, Jr., Manager, ACEE Project Office, NASA Langley Research Center

AIR TRANSPORTATION SYSTEMS AT THE END OF THE MILLENNIUM - Howard T. Wright, Director for Projects, NASA Langley Research Center

FUTURE COMMERCIAL VIABILITY OF COMPOSITES - Kenneth F. Holtby, Corporate Senior Vice President, The Boeing Company

"THE WAY AHEAD" - Russell H. Hopps, Vice President and General Manager, Engineering and Development, Lockheed-California Company

"COMPOSITE AIRCRAFT MATERIALS - THE FUTURE" - John B. DeVault, Vice President, Composite Materials and Structures, Hercules Aerospace

TUESDAY MORNING, AUGUST 14, 1984

SESSION 2: ACEE SECONDARY AND MEDIUM PRIMARY COMPOSITE STRUCTURES - STATUS REPORT

SESSION CHAIRMAN: Andrew J. Chapman, Technical Manager, Composites, ACEE Project Office, NASA Langley Research Center

[2]737, 757 and 767 COMPONENTS - John T. Quinlivan, Boeing Commercial Airplane Company

[3]DAMAGE TOLERANCE AND FAILSAFE TESTING OF THE DC-10 COMPOSITE VERTICAL STABILIZER - John M. Palmer, Jr., Clive O. Stephens, and Jason O. Sutton, Douglas Aircraft Company

[4]INITIAL STRENGTH AND HYGROTHERMAL RESPONSE OF L-1011 VERTICAL FIN COMPONENTS - Anthony C. Jackson, Lockheed-California Company

[5]RESIDUAL-STRENGTH TESTS OF L-1011 VERTICAL FIN COMPONENTS AFTER 10 AND 20 YEARS OF SIMULATED FLIGHT SERVICE - Osvaldo F. Lopez, NASA Langley Research Center

[5]WORLDWIDE FLIGHT AND GROUND BASED EXPOSURE OF COMPOSITE MATERIALS - H. Benson Dexter, NASA Langley Research Center, and Donald J. Baker, U. S. Army Structures Laboratory, NASA Langley Research Center

---

Footnotes on page x.

TUESDAY MORNING, AUGUST 14, 1984

[5]COMPARISON OF TOUGHENED COMPOSITE LAMINATES USING NASA STANDARD DAMAGE TOLERANCE TESTS - Jerry G. Williams, NASA Langley Research Center, T. Kevin O'Brien, U.S. Army Structures Laboratory, NASA Langley Research Center, and Andrew J. Chapman, NASA Langley Research Center

TUESDAY AFTERNOON, AUGUST 14, 1984

SESSION 3: REVIEW OF SELECTED NASA RESEARCH IN COMPOSITE MATERIALS AND STRUCTURES

SESSION CHAIRMAN: James H. Starnes, Jr., Head, Structural Mechanics Branch, NASA Langley Research Center

[5]SYNTHESIS AND TOUGHNESS PROPERTIES OF RESINS AND COMPOSITES - Norman J. Johnston, NASA Langley Research Center

[5]TENSILE STRENGTH OF COMPOSITE SHEETS WITH UNIDIRECTIONAL STRINGERS AND CRACK-LIKE DAMAGE - Clarence C. Poe, Jr., NASA Langley Research Center

[5]IMPACT DYNAMICS RESEARCH ON COMPOSITE TRANSPORT STRUCTURES - Huey D. Carden, NASA Langley Research Center

[5]POSTBUCKLING BEHAVIOR OF GRAPHITE/EPOXY PANELS - James H. Starnes, Jr., NASA Langley Research Center; John N. Dixon, Lockheed-Georgia Company; Marshall Rouse, NASA Langley Research Center

[1]DAMAGE TOLERANCE RESEARCH ON COMPOSITE COMPRESSION PANELS - Jerry G. Williams, NASA Langley Research Center

[5]STUDIES OF NOISE TRANSMISSION IN ADVANCED COMPOSITE MATERIAL STRUCTURES - Louis A. Roussos, Michael C. McGary, and Clemans A. Powell, NASA Langley Research Center

WEDNESDAY MORNING, AUGUST 15, 1984

SESSION 4: ACEE WING KEY TECHNOLOGIES

SESSION CHAIRMAN: Marvin B. Dow, Technical Manager, Composites, ACEE Project Office, NASA Langley Research Center

[2]COMPOSITE WING PANEL DURABILITY AND DAMAGE TOLERANCE TECHNOLOGY DEVELOPMENT - Robert D. Wilson, Boeing Commercial Airplane Company

[2]DESIGN DEVELOPMENT OF HEAVILY LOADED WING PANELS - Peter J. Smith, Boeing Commercial Airplane Company

---

Footnotes on page x.

WEDNESDAY MORNING, AUGUST 15, 1984

[3]THEORY AND ANALYSIS FOR OPTIMIZATION OF COMPOSITE MULTI-ROW BOLTED JOINTS - L. John Hart-Smith, Douglas Aircraft Company

[3]DESIGN AND TEST OF LARGE WING JOINT DEMONSTRATION COMPONENTS - Bruce L. Bunin, Douglas Aircraft Company

[4]COMPOSITE WING FUEL CONTAINMENT AND DAMAGE TOLERANCE TECHNOLOGY DEVELOPMENT - Charles F. Griffin, Lockheed-California Company

[4]COMPOSITE WING FUEL CONTAINMENT AND DAMAGE TOLERANCE TECHNOLOGY DEMONSTRATION - Tom W. Anderson, Lockheed-California Company

WEDNESDAY AFTERNOON, AUGUST 15, 1984

SESSION 5: REVIEW OF SELECTED DOD PROGRAMS

SESSION CHAIRMAN: H. Benson Dexter, NASA Langley Research Center

[6]MANUFACTURING TECHNOLOGY FOR LARGE AIRCRAFT COMPOSITE PRIMARY STRUCTURE (FUSELAGE) - DESIGN SELECTION - Hank R. Fenbert, Boeing Commercial Airplane Company; Harry S. Reinert, U.S. Air Force, MLTN, Wright-Patterson AFB, Vere S. Thompson, Boeing Commercial Airplane Company

[6]MANUFACTURING TECHNOLOGY FOR LARGE AIRCRAFT COMPOSITE WING STRUCTURE - Melvin A. Price, North American Aircraft Operations, Rockwell International Corporation, and D. R. Beeler, U.S. Air Force, AFWAL/MLTN, Wright-Patterson AFB

[6]MANUFACTURING TECHNOLOGY FOR LARGE COMPOSITE FUSELAGE STRUCTURE - Richard L. Circle and R. Dennis O'Brien, Lockheed-Georgia Company

[6]DAMAGE TOLERANCE OF COMPOSITES - John E. McCarty, Boeing Military Airplane Company

[1]ADVANCED COMPOSITE AIRFRAME PROGRAM (ACAP) - Tom Mazza, U.S. Army Applied Technology Laboratory, Fort Eustis

[6]COMPOSITE STRUCTURES - IMPROVED DESIGNS FOR MILITARY AIRCRAFT - Anthony Manno, Mark S. Libeskind, Ramon Garcia, and Edward F. Kautz, U.S. Navy, Naval Air Development Center

[1]COMPOSITE STRUCTURES IN THE JVX AIRCRAFT - Keith Stevenson, Bell Helicopter Corporation

---

Footnotes on page x.

THURSDAY MORNING, AUGUST 16, 1984

SESSION 6: ACEE ADVANCED COMPOSITE STRUCTURES TECHNOLOGY

SESSION CHAIRMAN: Jon S. Pyle, Technical Manager, Composites, ACEE Project Office, NASA Langley Research Center

[2]BOEING - PRESSURE CONTAINMENT AND DAMAGE TOLERANCE IN FUSELAGE STRUCTURE - Ronald W. Johnson, Boeing Commercial Airplane Company

[3]DOUGLAS - JOINTS AND CUTOUTS IN FUSELAGE STRUCTURE - D. Joseph Watts, Douglas Aircraft Company

[4]LOCKHEED - IMPACT DYNAMICS AND ACOUSTIC TRANSMISSION IN FUSELAGE STRUCTURE - Anthony C. Jackson, Lockheed-California Company

[4]LOCKHEED COMPOSITE TRANSPORT WING TECHNOLOGY DEVELOPMENT - PROGRAM OVERVIEW

COVER/RIB CONCEPTS - Arthur M. James, Lockheed-California Company

SPAR/ASSEMBLY CONCEPTS - William E. Harvill, Jr., Lockheed-Georgia Company

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[1]Oral presentations only.
[2]Papers contained in NASA CR-172358.
[3]Papers contained in NASA CR-172359.
[4]Papers contained in NASA CR-172360.
[5]Papers contained in NASA CP-2321.
[6]Papers contained in NASA CP-2322.

INITIAL STRENGTH AND HYGROTHERMAL RESPONSE OF L-1011 VERTICAL FIN COMPONENTS

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ACEE Composite Structures Technology Conference
Third Special Oral Review
August 13-16, 1984

INTRODUCTION

This paper summarizes the results of the "Production Readiness Verification Tests (PRVT)". These tests were performed as part of the Advanced Composite Vertical Fin for L-1011 Aircraft (ACVF) program NAS1-14000.

The initial program plan included flight service, with periodic inspections. It was anticipated that airlines would commit aircraft which are used in routine operations to the program, for the purpose of obtaining that flight service evaluation. Although a considerable effort was expended in persuading airline companies to participate in the development program, it was soon recognized that the idea of evaluating composite primary structure on passenger carrying aircraft was not practical at that time. A more achievable program goal was to generate technology that would provide the confidence needed to commit the use of advanced composite materials for primary structures of future aircraft. Therefore, the ACVF program was restructured in order to accomplish the new program goals.

A phase added to the program in the restructuring was Phase III, Production Readiness Verification Tests (PRVT). These tests were designed to provide information to answer the following questions:

- What is the range of production qualities that can be expected for components manufactured under conditions similar to those expected in production, and how realistic and effective are proposed quality levels and quality control procedures?
- What variability in static strength can be expected for production quality components, and are the margins sufficient to account for this variability?
- Will production quality components survive extended time laboratory fatigue tests involving both load and environment simulation of sufficient duration and severity to provide confidence in in-service durability?

To provide data, 22 components of each of two key structural elements of the ACVF were fabricated for test. One element represented the front spar/fuselage attachment area, and the other element represented the cover/fuselage joint area. Ten of each element were static strength tested. Six of each element were durability tested for the equivalent of ten years of service and statically tested at NASA Langley Research Center to determine their residual strengths. The remaining six of each were durability tested for the equivalent of 20 years of service. Two of each of these last six were durability tested at strain levels 1.5 times those in the basic program. At the completion of 20 years the remaining specimens were statically tested at NASA Langley Research Center to determine their residual strengths.

PRODUCTION READINESS VERIFICATION TEST OBJECTIVES

When the Advanced Composite Vertical Fin program was restructed in 1976, it was realized that flight service evaluation even if possible would not provide the data for evaluating production readiness for medium primary composite structures in an acceptable time frame. The three areas which needed addressing are shown in Figure 1 and are expanded upon below as questions.

1. What is the range of production qualities that can be expected for components manufactured under conditions similar to those expected in production, and how realistic and effective are proposed quality standards and quality control procedures?
2. What variability in static strength can be expected for production quality components, and are the design allowables sufficient to account for this variability?
3. Will production quality components survive laboratory fatigue tests involving both load and environment simulation of sufficient duration and severity to provide confidence in long-term durability in the service environment?

The selected components were fabricated during 1978 and early 1979. Static testing was completed the spring of 1979.

Cyclic durability testing commenced on May 3, 1979 with spars and on June 18, 1979 with covers. The durability testing was completed in September of 1983.

After the completion of testing the components were shipped to NASA Langley Research Center for residual strength testing.

- **DEMONSTRATE REPRODUCIBILITY**
- **DETERMINE VARIABILITY IN STATIC STRENGTH**
- **DEMONSTRATE LONG-TERM DURABILITY**

Figure 1

FIN STRUCTURAL CONFIGURATION

The structural configuration is illustrated in Figure 2. The covers are one piece cocured skins and closed hat stiffeners and are designed primarily by stiffness. The skin tapers in steps from 34 plies at the root end to 16, 14, then 10 at the tip end. The edges are built up to 0.12 inch (24 plies) to allow for countersinking holes without feather edges. The closed hat section stiffener was selected because of its torsional stability and the fact that it did not have to be tied to each rib. The stiffener is built up of two five-ply segments with a ten-ply segment sandwiched between them in the crown.

The eleven ribs fall into three basic categories: The two lower ribs are actuator ribs, the next six are truss ribs, and the upper three are solid web ribs.

The actuator ribs consist of a partial solid graphite web at VSS 90.19 and a combination solid graphite web and graphite cap, aluminum truss rib at VSS 97.19. The solid web is a 16-ply layup. The sides adjacent to the covers are flanged to provide part of the skin attachment. Additional cap is provided by a C-section consisting of a 19-ply layup. The forward portion of this rib consists of the graphite-epoxy C-section caps and aluminum cruciform extruded truss members. The truss rib caps are C-section caps consisting of 19 plies. The truss members are again aluminum cruciform extrusions.

The solid web ribs are a sandwich design using a syntactic epoxy core. Syntactic epoxy is an epoxy system filled with glass microballoons which has about half the density of graphite epoxy.

The front and rear spars are similar in shape and size and are basically one-piece components with rib attach angles, stiffeners, caps, and webs integrally molded in a single cocured operation.

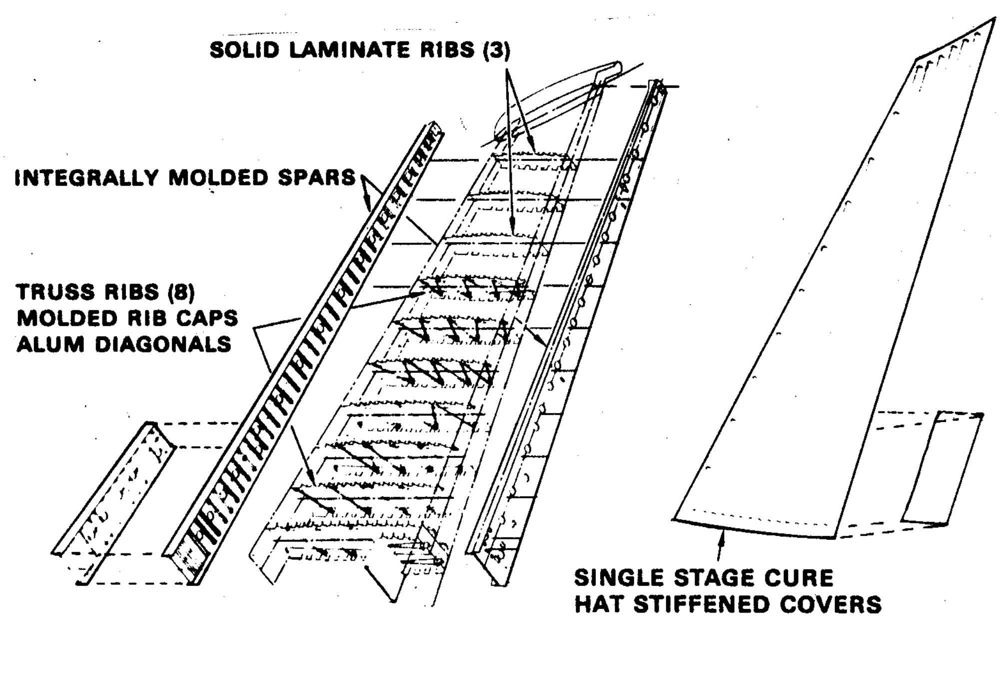


Figure 2

TEST COMPONENTS

The cover specimen consisted of a section of the root end near the rear spar. It contained three stiffeners which taper out into the root joint area. The stiffener spacing was 7.2 inches and the basic skin was 16 plies or approximately 0.08 inch thick. The root end skin is built up to 34 plies or 0.17 inch thick to mate with the double lap splice joint to the L-1011 afterbody structure. A left and a right hand component are shown in Figure 3.

The spar specimen consisted of the lower 72 inches of the front spar, as shown in Figure 3. It contained integral blade stiffeners. Access holes were machined in the web in alternate bays. There was no reinforcing around the access holes. The uppermost access hole was omitted because of the local high load introduction in the test setup.

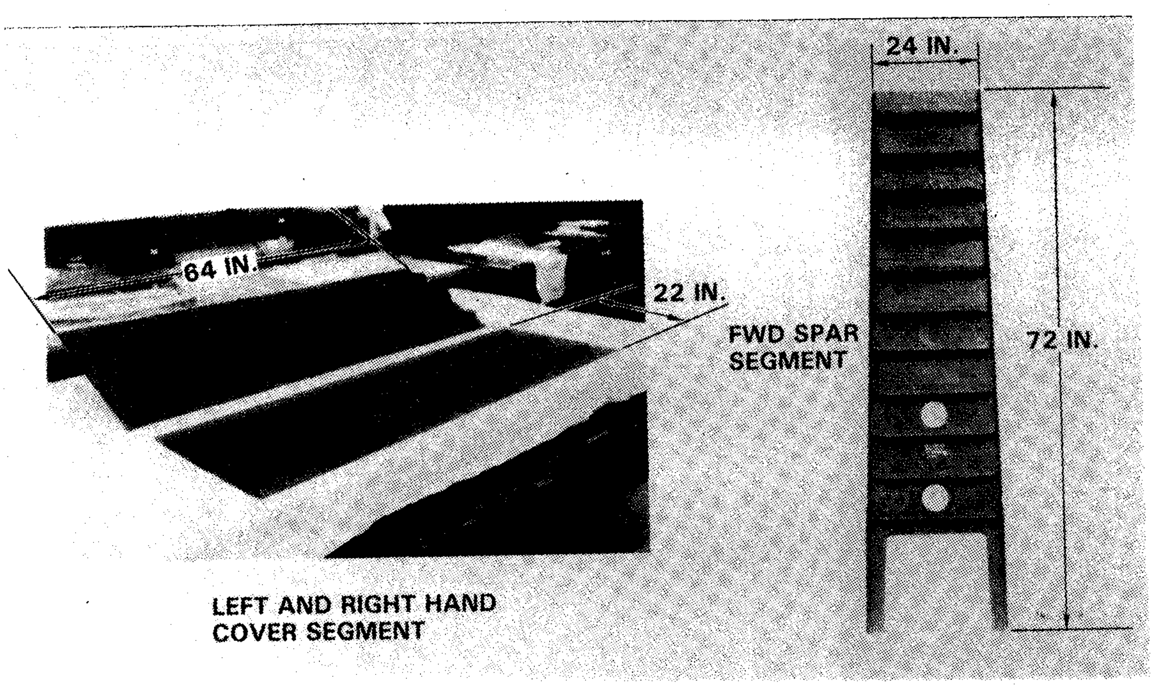


Figure 3

REPRODUCIBILITY

The PRVT cover components were fabricated in the Calac plastics production shop by production personnel with manufacturing research assistance. The PRVT spar components were fabricated in the Gelac manufacturing research shop using both production personnel and manufacturing research personnel. Twenty-eight cover components and twenty-four spar components were produced. Inspections showed that twenty-two of the cover components were acceptable for test and that all the spar components were acceptable as summarized in Figure 4. Much of the cause for the scrap of six covers was due to tooling problems which resulted in lack of pressure in some critical areas and caused porosity. One cover was scrapped because of a machining error and one because of foreign matter, which was suspected to be backing paper. A review of the inspection results shows that the ultrasonic inspection techniques successfully screened components containing porosity, voids and foreign matter.

Thus the answer to question one is as follows:

The range of production qualities that can be expected for components manufactured under conditions similar to those expected in production has been established. The spars were produced using tooling which underwent only minor modifications during the run of 24 components, similar to a production run. The covers were fabricated using tooling that underwent various modifications during a run of 28 components. Thus extremes of the production environment were encountered.

The quality control procedures used proved adequate in identifying discrepancies. In particular NDI techniques developed and refined during the program worked very well.

<u>COVERS</u> — 28 FABRICATED

4 SCRAPPED DUE TO TOOLING PROBLEM
1 SCRAPPED DUE TO MACHINING ERROR
1 SCRAPPED DUE TO BACKING PAPER IN LAMINATE

<u>SPARS</u> — 24 FABRICATED

NONE SCRAPPED

Figure 4

TEST COMPONENT DEFECTS

During fabrication of the test components, various defects occurred (see Figure 5). Some of these were within the acceptable range and some were severe enough to cause rejection. However, some defects were marginal or repairable. In order to establish the validity of the accept/reject criteria, marginal components were accepted for test. If two components had similar defects every effort was made to assign one to static testing and one to durability testing. High resin content occurred in several spars and low resin content occurred in one cover and two spars. Isolated areas of porosity occurred in some covers and some spars. Mark-off from the tooling was a recurring problem for both covers and spars. Minor delaminations occurred in several spars and were repaired. Two spars were damaged in shipment and major repair for delaminated stiffeners was necessary. One cover had masking tape cured in a hat crown. During machining and handling, various cover components suffered minor chipping and scuffing damage.

During test assembly and set-up, some problems were encountered with hole drilling resulting in elongation or backside breakout.

None of these defects had any effect on durability or measurable effect on static strengths.

- **HIGH RESIN CONTENT** — **36% BY WEIGHT**
- **LOW RESIN CONTENT** — **25% BY WEIGHT**
- **POROSITY**
- **MARK-OFF**
- **DELAMINATION** — **REPAIRED**
- **MASKING TAPE IN LAMINATE**
- **CHIPPING AND SCUFFING**
- **DRILLING** — **ELONGATED HOLES** — **BREAKOUT**

Figure 5

COVER STATIC TEST RESULTS

The results of the ten cover static tests are summarized in Figure 6. The results show excellent uniformity. The highest failure occurred at 167.4 percent of Design Ultimate Load (DUL) and the lowest at 153.9 percent DUL. The mean failure load was 160 percent DUL. The DUL was 57,500 pounds.

An analysis of component was made prior to test to predict the failure load. This analysis was performed using average strength data from coupon testing. The predicted failure load proved to be conservative thus demonstrating that the allowables are sufficient to account for material and component variability.

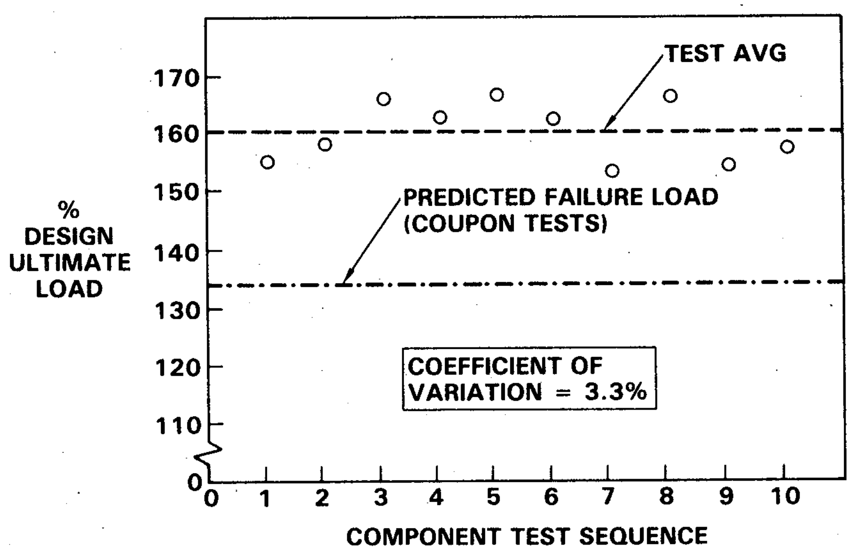


Figure 6

SPAR STATIC TEST RESULTS

The results of the ten spar static tests are summarized in Figure 7. As with the covers, the results show excellent uniformity but do have a higher coefficient of variation. This is no doubt due to the fact that the spars were tested under bending loads and the covers were tested under uniaxial compression loads. The bending induces more complex internal load reactions. The highest failure occurred at 149.4 percent DUL while the lowest occurred at 124.8 percent DUL. The mean failure load was 134.9 percent DUL. The DUL was 20,715 pounds at the upper jack.

An analysis of the component was made prior to testing to predict the failure load. This analysis was also performed using average strength data from coupon testing. The predicted failure loads proved to be conservative thus demonstrating that the allowables are sufficient to account for material and component variability.

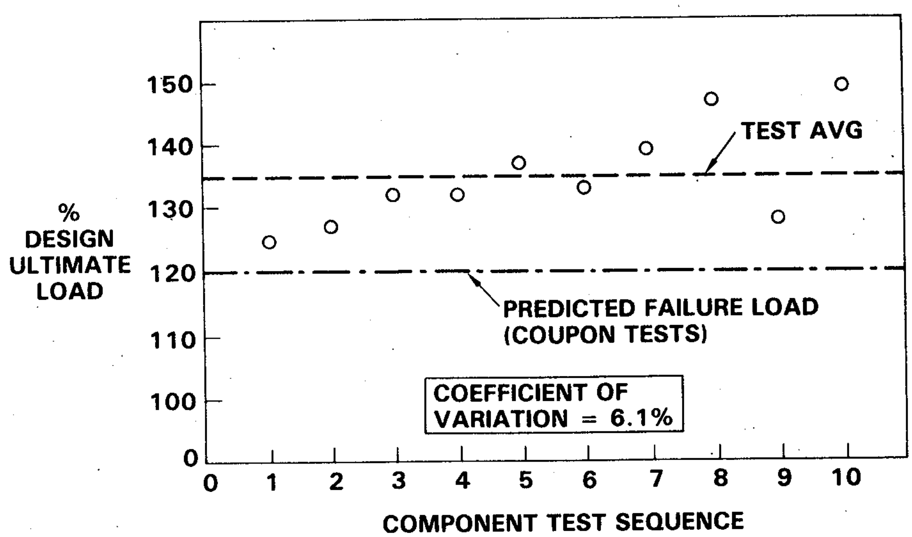


Figure 7

COEFFICIENT OF VARIATION IN STATIC STRENGTH OF SOME STRUCTURAL MATERIALS

The static test results showed excellent uniformity. The coefficients of variation (CV) compare favorably with those of other common structural materials as shown in Figure 8. The allowables used were derived from coupon data and the CVs of some of these data are shown in Figure 8 also. The failure modes of the covers and spars are influenced primarily by stiffness. The specimens in all cases failed at loads higher than predicted. The allowables used for prediction were based on average coupon data whereas design allowables are statistically reduced below those levels. The allowable thus proved adequate to account for structure static variability.

MATERIAL	COMPONENT	NO. SPEC.	LOADING	COEFFICIENT OF VARIATION PERCENT
GRAPHITE-EPOXY	PRVT-COVER	10	COMPRESSION	3.3
GRAPHITE-EPOXY	PRVT-SPAR	10	BENDING	6.1
GRAPHITE-EPOXY	SPOILER	15	BENDING	6.6
GRAPHITE-EPOXY	LAMINATE COUPONS	411	TENSION	5.7
GRAPHITE-EPOXY	LAMINATE COUPONS	411	TEN-MODULUS	4.0
GRAPHITE-EPOXY	LAMINATE COUPONS	290	COMPRESSION	9.0
GRAPHITE-EPOXY	LAMINATE COUPONS	290	COMPR-MODULUS	5.2
WOOD	MOSQUITO WINGS	5	BENDING	10.3
WOOD	PLYWOOD SHEAR WALL	27	SHEAR	9.7
CONCRETE	TEST CYLINDERS	216	COMPRESSION	10.6
ALUMINUM	7049-T73 DIE FORGING	384	TENSION	3.2
ALUMINUM	A357-T6 CASTING	804	TENSION	5.5
TITANIUM	TI-5AL-2.5SN SHEET	565	TENSION	3.9
STEEL	STRUCTURAL STEEL	3982	TENSION	7.1
STEEL	17-7PH SHEET	88	TENSION	5.1

Figure 8

COVER TEST SETUP

The test setup is shown in Figure 9. A rigid steel reaction frame was used to stabilize the test cover during compression loading. The cover was prevented from buckling at the two rib supports and at the root end tee through the use of three 27-inch long aluminum alloy flexure plates. These plates were designed to provide a restraint coefficient of approximately 1.0 at the test panel. Kick loads were reacted through four solid steel links attached to the ends of the cover assembly (at the centroids) and to the reaction frame.

The test cover was installed, in an upside-down position, in the 400 kip Universal static test machine. It was centered between the lower compression plate (resting on the movable base of the machine) and a rigid compression head attached to the fixed upper end of the machine. The reaction frame rested on the movable base of the test machine and was free to move upward along with the loading head.

In the installation procedure, the upper compression head was adjusted (through shimming) until its lower surface was parallel to the upper surface of the lower compression plate. This parallelism was later demonstrated by loading the test cover to 30 kips then observing the head deflections measured at four symmetrically located points near the upper compression head. The two edges of the cover were supported by split tube clamps with adjustable slots.

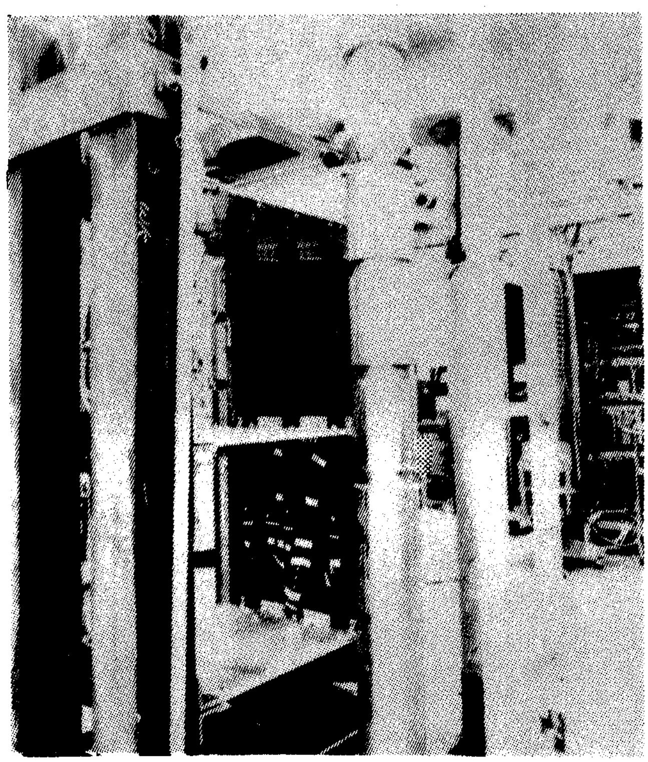


Figure 9

TYPICAL COVER STATIC FAILURE

The covers all behaved in a similar manner up to failure. All but one cover failed in the same manner. Figure 10 shows a typical failure from the skin side. The failures were initiated by the skin buckling between the hats and the buildup of interlaminar tension stresses that caused the skin and hats to separate. These failures occurred in the 16-ply skin between the two rib supports.

A nontypical failure occurred in test 3 (cover no. 17). The failure occurred between the root end and the first rib support, very close to the rib in the last of the 16-ply area.

High speed movies (400 frames/second) were taken during each failure run. In most cases the failures occurred within 1/400th of a second and were consequently not picked up by the two high speed cameras. In test 4, cover no. 7, the camera did show the failure initiation at the center hat.

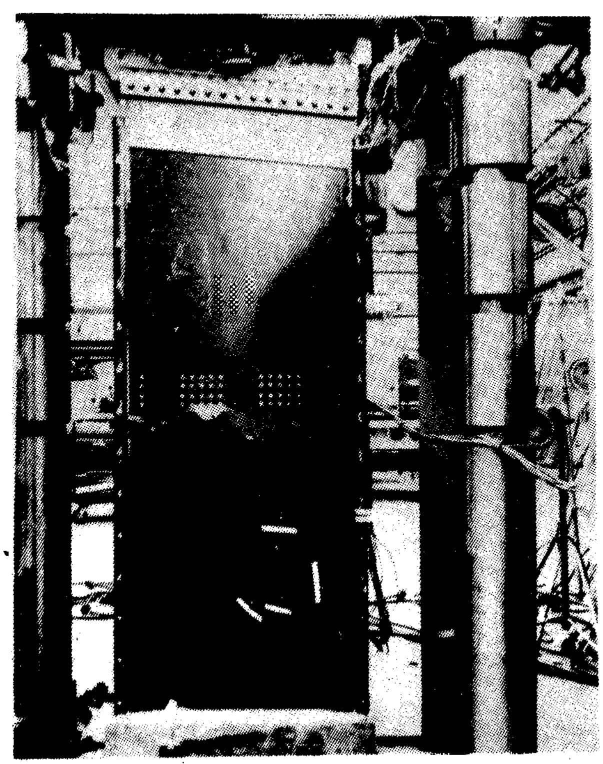


Figure 10

SPAR TEST SETUP

Aluminum plates were bolted to the caps to simulate the covers. The covers locally have a modulus roughly equivalent to aluminum. The plates provided the correct balance of axial load in the spar caps and shear in the spar webs. A typical test setup is shown in Figure 11. The metallic structure shown at the top of the spar is test structure to help introduce the loads.

Each spar was loaded in bending by two hydraulic jacks, one at the tip, and one at the lower rib intersection with the spar. The spar was stabilized at the loading points by steel rods.

The spar was mounted vertically and cantilevered off a rigid I-beam attached to the floor. The loading jacks were attached to a vertical I-beam which was part of a larger general purpose test reaction frame.

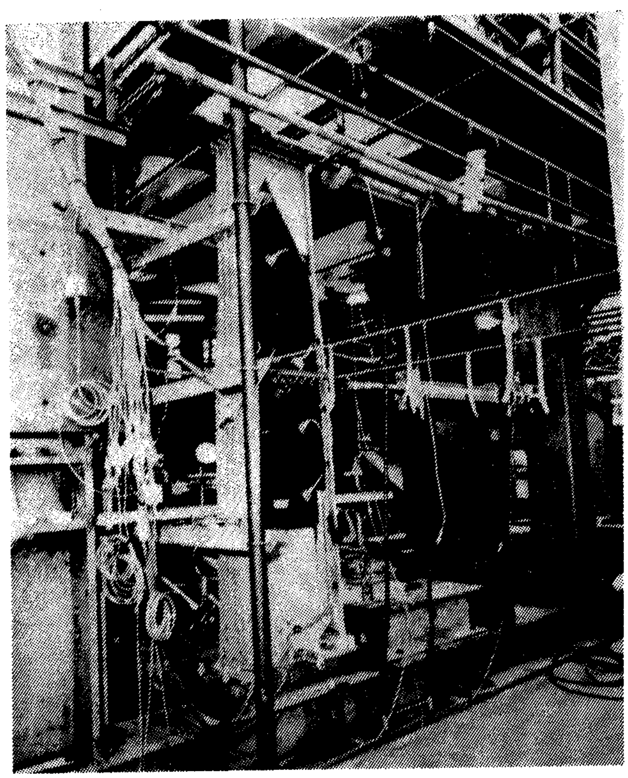


Figure 11

TYPICAL SPAR FAILURE

The spars all behaved in a similar manner up to failure. All failures, except one, were in the predicted bay through the access hole, at specimen Sta 32.45, and were similar in appearance. Figure 12 shows a typical failure from the aft side of the spar. It can be seen in Figure 12 that the stiffener just above the access hole at which failure occurred popped off. The failure shown near the top of the specimen was a secondary failure caused by the high deflection after the primary failure.

Spar no. 1 was the nontypical failure. A large buckle occurred at the lower access hole at specimen station 19.89 just prior to failure. Failure occurred at the lower access hole.

An anomaly occurred during the first test, spar no. 14. Two test runs to design ultimate were made with satisfactory results. The failure run was terminated at 123 percent of design ultimate when it was noted that delamination of the web had occurred. This delamination was at the second access hole at specimen station 32.45. The delamination was not visible when the load was removed. The spar was then reloaded to design limit, held for 30 seconds and then unloaded. After a review of the data, it was decided to retest the spar to determine the maximum load capability after the initial failure (or delamination of the spar web). The spar was reloaded to design limit and held while a photograph of the delamination was taken. Loading was then continued to rupture, which occurred at 125.0 percent of design ultimate.

The high speed movies revealed little that was not observed visually.

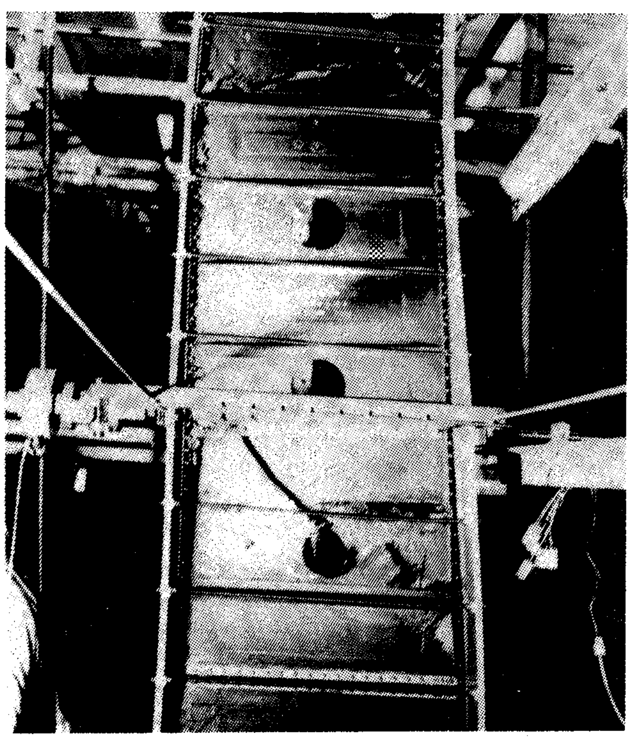


Figure 12

DURABILITY TEST OBJECTIVES

The durability testing was designed to give quasi-realtime results and to bridge effectively the gap between accelerated coupon testing which is completed in a matter of a few weeks and the real-time exposure of structural components in flight service.

The normal ground/air/ground environment causes both absorption and desorption of moisture by the epoxy matrix. This causes a laminate to swell and shrink in thickness. This effect would be most detrimental in joint areas. The continuous swelling and shrinking may loosen the joint over a period of time or cause other detrimental effects. This would not only be an undue maintenance burden but might lead to structural failures. Because of potential galvanic corrosion problems when graphite and aluminum are in contact the environmental cycling would validate the corrosion protection systems incorporated. Several specimens had some form or repair and durability of these repairs would also be validated.

In order to determine any degradation overall stiffness would be monitored by recording loads and deflections and local strains would be measured by strain gages. Moisture weight gain would be determined by periodic weighing of traveller coupons which were distributed in the chambers.

Figure 13 summarizes the objectives and the data monitored during the testing.

DETERMINE EFFECTS ON LONG-TERM DURABILITY OF:

- **FLIGHT BY FLIGHT LOADING WITH GAG-TEMPERATURE/HUMIDITY CYCLING**
- **OPERATING STRAIN LEVELS**
- **ALUMINUM/GRAPHITE-EPOXY INTERFACES**
- **JOINTS**
- **REPAIRS**

MONITOR:

- **OVERALL STIFFNESS**
- **LOCAL STRAIN**
- **MOISTURE WEIGHT GAIN**

Figure 13

PRVT DURABILITY TESTS
LOADS/THERMAL CYCLE SEQUENCE

Figure 14 shows a schematic of the durability test profile.

The thermal cycles used in the test represent about 20 percent of the total cycles expected in the fin life-time. The ambient temperature, then, was assumed to be that exceeded on the average 20 percent of the time, or an ambient of 80-85°F, based on National Weather Service temperature exceedance data. This ambient temperature range converts to a skin temperature of about 140°F if a painted fin is assumed with the darker color of paint predominating. The lower bound of temperatures, -30°F, was selected to be certain that the moisture in the laminate will be fully frozen prior to beginning the heating cycle.

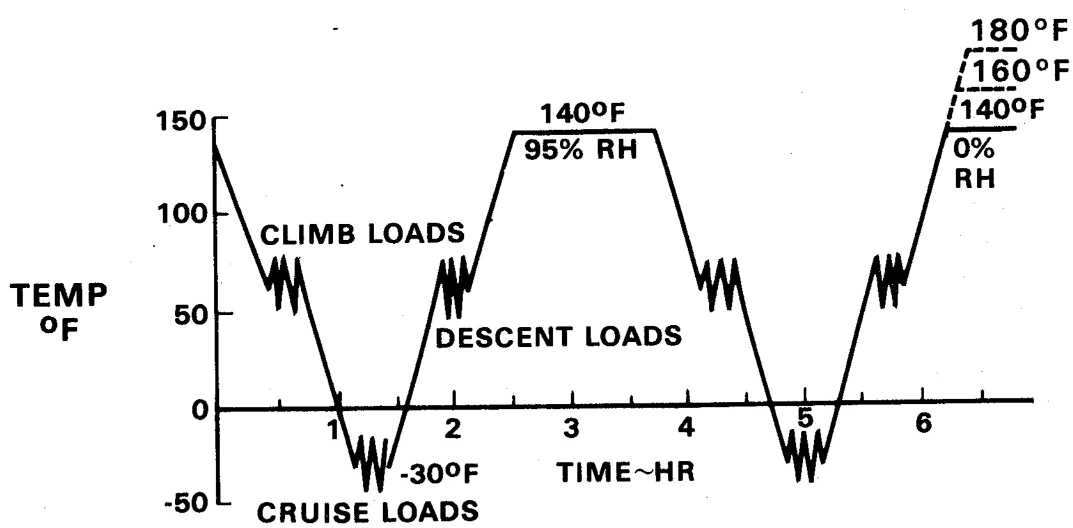
The relative humidities were selected at 0 percent and 95 percent. This represents fairly typical conditions found in Las Vegas and Miami, respectively, in summer months. The flight cycle is thus typical of continuous operation between the two cities and would exercise the moisture gradients between layers of the laminate to a maximum.

In addition to the environmental spectrum, at dispersed times during the test the temperature was allowed to reach 160° (40 times), and 180° (10 times). These latter conditions simulated the infrequent maxima expected in service.

The environmental spectrum was intended to accomplish two primary objectives, namely to provide large changes in moisture content through the plies of the laminate and to produce some acceleration of the testing program, simulating a higher number of equivalent cycles than the actual 5800.

One lifetime represents 36,000 flights, the equivalent of 20 years of service.

Loading cycles were applied in the climb, cruise, and descent phases of a flight.



LOADS FOR 36,000 FLIGHTS TRUNCATED
INTO 5,800 THERMAL CYCLES

Figure 14

COVER DURABILITY TEST ASSEMBLY

The cover specimens were assembled into units consisting of one left hand and one right hand component as shown in Figure 15. Each component was supported at rib locations by flexure plates representing the buckling restraint of the ribs. These flexure plates were mounted off the shear beam assemblies so that the flexure for one cover did not connect with or influence that for the other cover. The setup was designed so that each cover was loaded individually by one jack so failure of one cover would not be expected to cause failure of the other. The edges of the cover components were restrained from free edge buckling by metal angles which allowed the part to slide in plane but prevented out-of-plane deflections.

The root joint was made through a representative joint in the composite to a specially designed test part which picked up with the load jacks. The other end of the panel was reinforced with fiberglas, and metal plates bolted to the component to pick up the angles which attached the assembly to the load reaction frame.

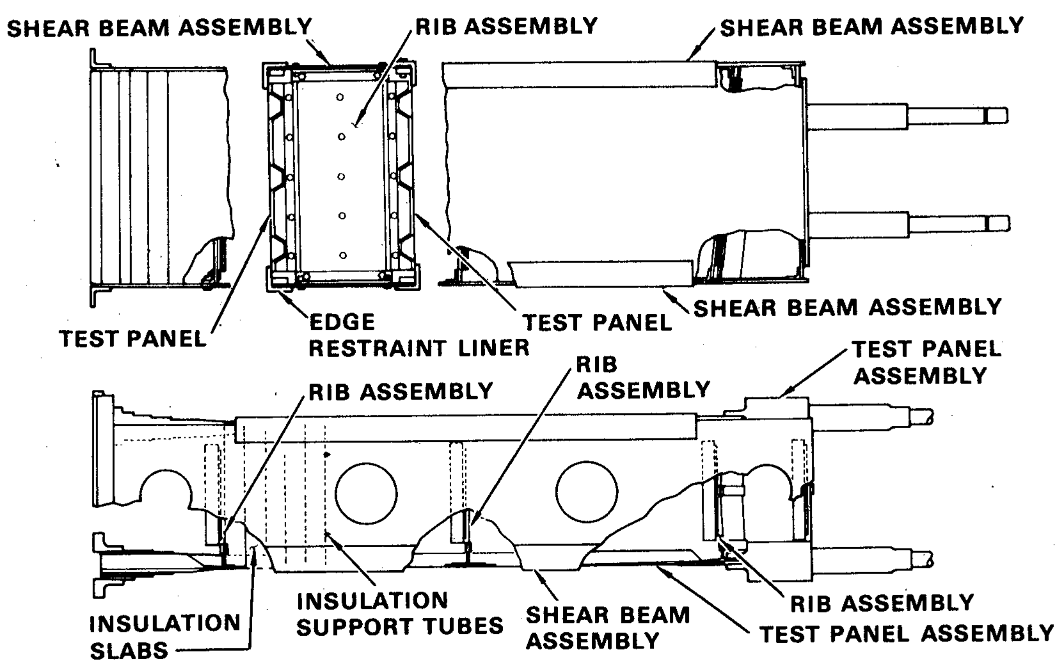


Figure 15

SPAR DURABILITY TEST ASSEMBLY

A decision was made to load the durability spars with one jack at the tip instead of one at the tip and one at a rib location further down the spar as in the static test. This decision was based on the difficulty in controlling the two jacks to operate together over a long period of time and the potential for breaking or damaging the part if one jack malfunctioned. The spars were also assembled in pairs as shown in Figure 16 but with one jack loading both spars so as to free up additional computer channels for instrumentation monitoring. The hazard was that failure of one spar would cause failure of the other spar. In order to obtain as close a match as possible with the required cap and web loads aluminum plates were attached to the spar caps to represent the effective cover.

The upper loading fixture was attached to the spars through metal doublers mechanically attached to the web and cap. The root joints were made to a base plate which would attach to the load reaction frame.

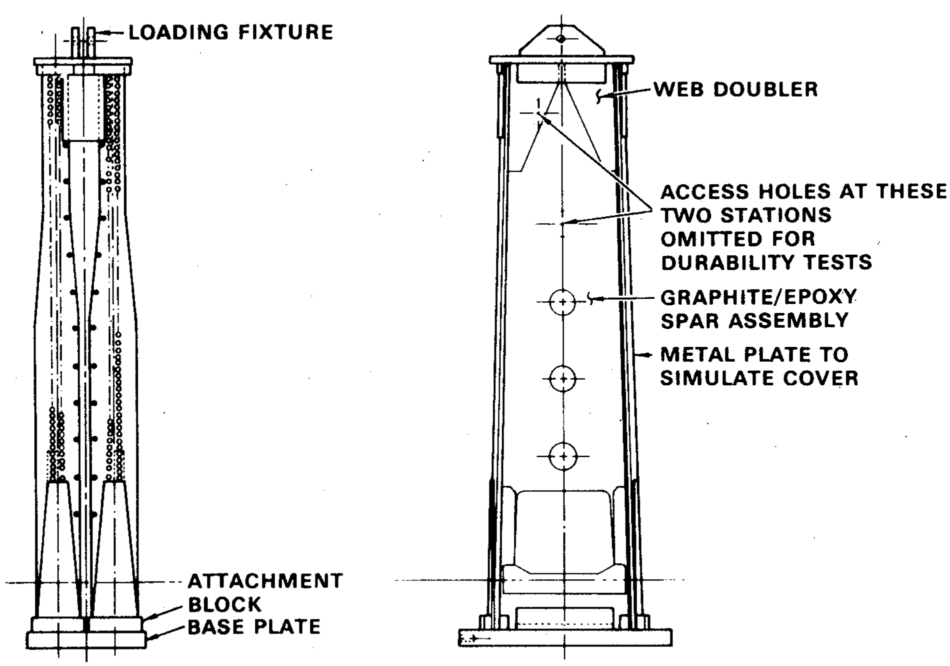


Figure 16

COVER TEST SETUP

The cover load reaction frame had a bolted-on top which was removed during specimen installation. A cutout in the chamber top allowed the three cover pairs to be lowered vertically into place. The four load rods on the bottom of each pair went through holes in the chamber bottom and bolted to a cross bar shown in Figure 17. This cross bar was attached to a load cell, then to the hydraulic jack. When all three pairs were in place, the top loading beam was replaced on the reaction frame structure and bolted securely. The covers were then brought up close to the upper beams by a series of 12 bolts per cover, Devcon potting material was applied to assure proper alignment, and allowed to cure before final tightening of the attachment bolts. The gap between the specimens and chamber access hole was sealed with closed-cell polyethylene foam.

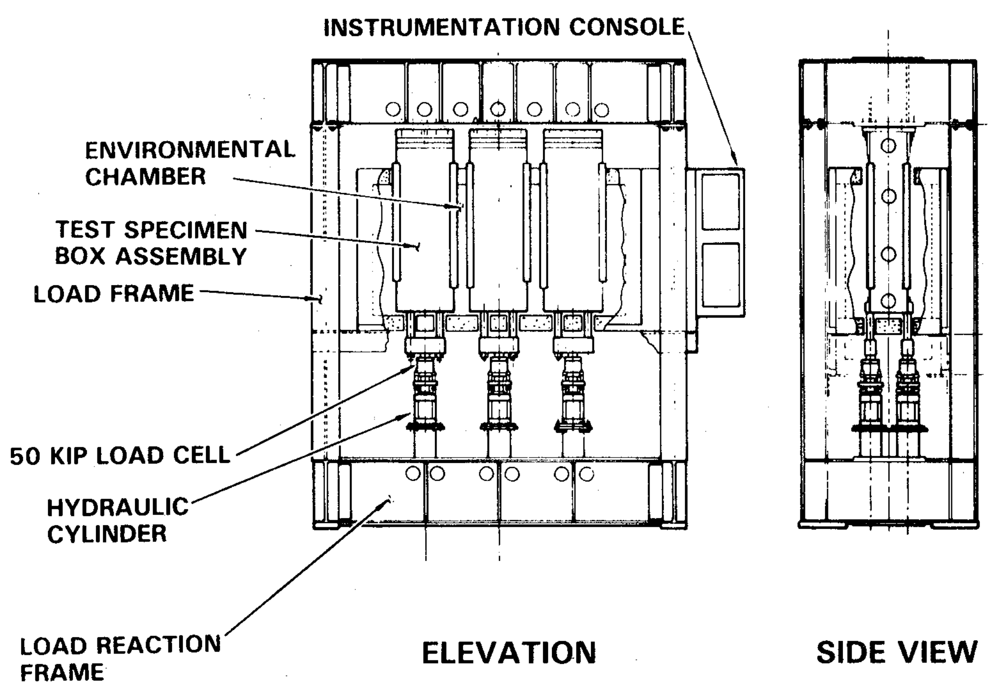


Figure 17

COVER CHAMBER

Based on the number of test components, size, thermal mass, loads, time restraints, available floor space, etc., an analysis of all pertinent factors determined that the optimum test facility would require two chambers with 90 inches long by 52.5 inches high by 32 inches wide internal working dimensions for the ten durability cover specimens, and two chambers 105 inches wide by 120 inches high by 40 inches deep for the ten durability spar specimens. A typical cover chamber is shown in Figure 18.

The chambers were constructed of a continuously heliarc welded series 304 stainless steel inner liner and an angle frame reinforced 16-gage cold rolled steel outer case insulated with Upjohn Company Trymer CPR 9945 modified isocyanurate cellular plastic. The cover chamber had double doors on the front and back sides permitting easy access for inspection of the specimens. The spar chamber had one large door on the front exposing the entire working volume. The doors were designed with both an inner and outer gasket to minimize water buildup in the gasket space and reduce thermal losses through the door breakers. The floor had drains for condensed moisture.

Air circulation within the workspace was accomplished by a blower system drawing air from the workspace, blowing it through heating and cooling coils, and returning it to the workspace.

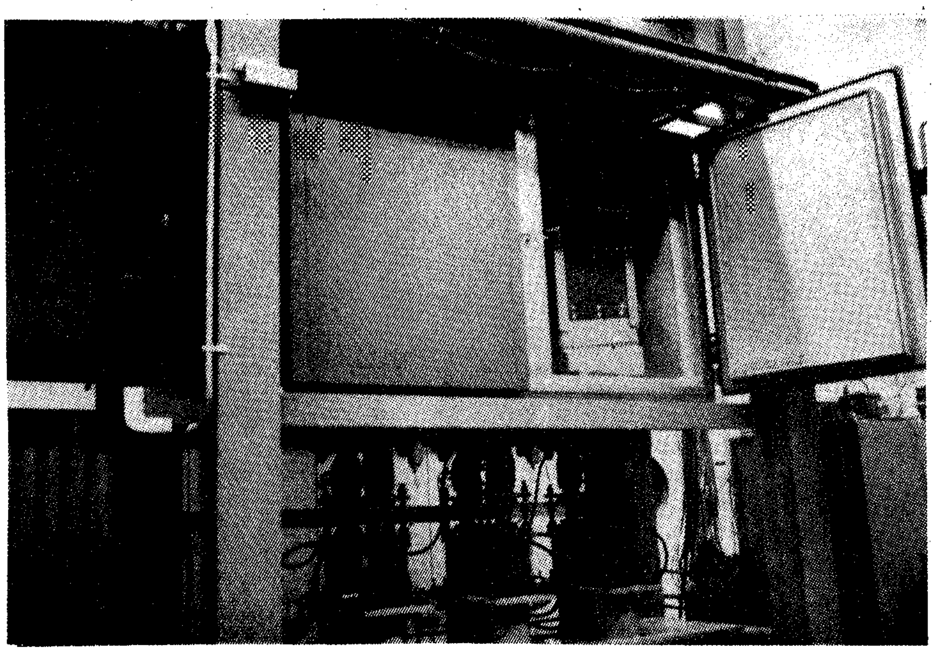


Figure 18

SPAR TEST SETUP

The load reaction frame was a welded steel structure and is shown in Figure 19. Three spar pairs were placed in each of the two chambers and were bolted through the chamber floor with rigid supports down to the load reaction frame base. A load rod extended through the aft wall of the chamber from the loading brackets on top of each spar pair and attached through a load cell and hydraulic jack to the load reaction frame.

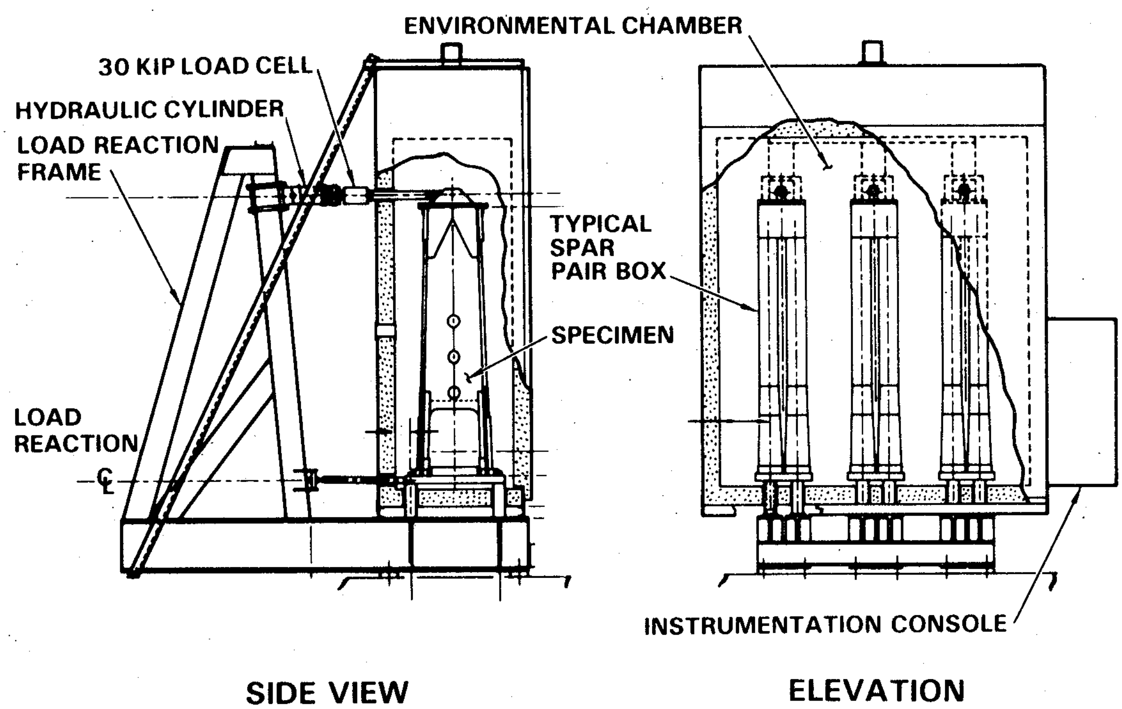


Figure 19

SPAR CHAMBER

A spar chamber is shown in Figure 20. The spar chambers were constructed in a similar manner to the cover chamber. One refrigeration system cooled all four chambers, thus, the cooling cycle was staggered into one-thirds so that only one spar or two cover chambers are cooling at any one time to minimize refrigeration capacity requirements. Should one chamber experience a down condition, it had to remain off until it could come in at the proper temperature cycle sequence.

A central steam generator was also contained in the refrigeration system machinery console to increase vapor content in the various chambers. The steam generator included: a sight glass; an automatic low water cut-out; an automatic water level control, and a pressure control relief valve. Steam was proportioned by the humidity controllers via a solenoid valve to each chamber. Just downstream of the inlet air orifice a one-inch pipe ran perpendicular to the air stream the full width of the air orifice and sprayed steam into the air stream through six 3/32-inch holes.

Chamber high temperature was achieved by using Nichrome element heaters controlled by heavy duty mercury relays integrated with a dry bulb temperature controller. The heaters were protected by a separate power controller interlocked with a high temperature safety thermostat and a solid state electronic high-low temperature safety control interlocked with the setup relay and visual and audible alarm, as well as the central computer monitoring system.

Figure 20

MOISTURE HISTORY

Moisture content in the spars and covers was tracked by installation of weight gain travelers in each chamber. A total of 50 travelers were used. In the spar chambers a tee section of cap and web material and a web section were cut from spar scraps so that the traveler had section properties the same as the spars. Likewise, cover hat and skin sections were fabricated.

These travelers were cut, weighed, and installed in the chambers prior to the test start-up, without any drying just as the spars and covers. Each one was indexed to a specific location in the chamber and after weighing returned to that spot. Weight gain measurements were made at intervals, the results are summarized in Figure 21.

In April 1982, twenty-seven travelers were removed, some from each chamber and type, and dried in a vacuum at 150°F until weight loss stabilized. These data were then used to adjust the previously measured weight gains to account for initial moisture content.

Fluctuations occurred in the periodic data. This was due to the point in the thermo/humidity cycle at which the coupons were removed for weighing and to malfunctions of the humidity generator. The spars leveled off at approximately 0.9 percent and the covers at 1 percent weight gain used for design allowables determination is adequate.

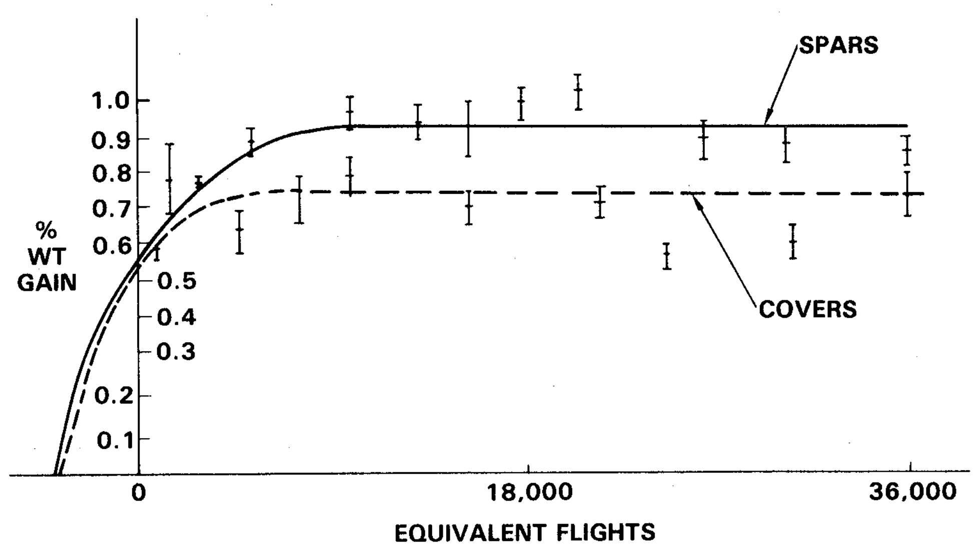


Figure 21

DEFLECTION HISTORY

Figure 22 shows relative deflections for the covers and spars measured at intervals during the testing.

The spar components exhibited a drop in deflection during the first 6000 flights after which the deflections remained essentially constant for any given load. This apparent stiffening of the components was due to swelling of the laminate as it absorbed moisture which in turn caused an increase in the clamping pressure by the fasteners producing increased frictional forces. The fastener holes were slightly oversize.

The cover components did not exhibit any changes in deflection. The fastener holes in the root joint were closely controlled to give minimum oversize and overall deflection was only one tenth that of the spar components. The lower deflections are due to the fact that the spar was loaded in beam bending while the cover was loaded in axial tension and compression.

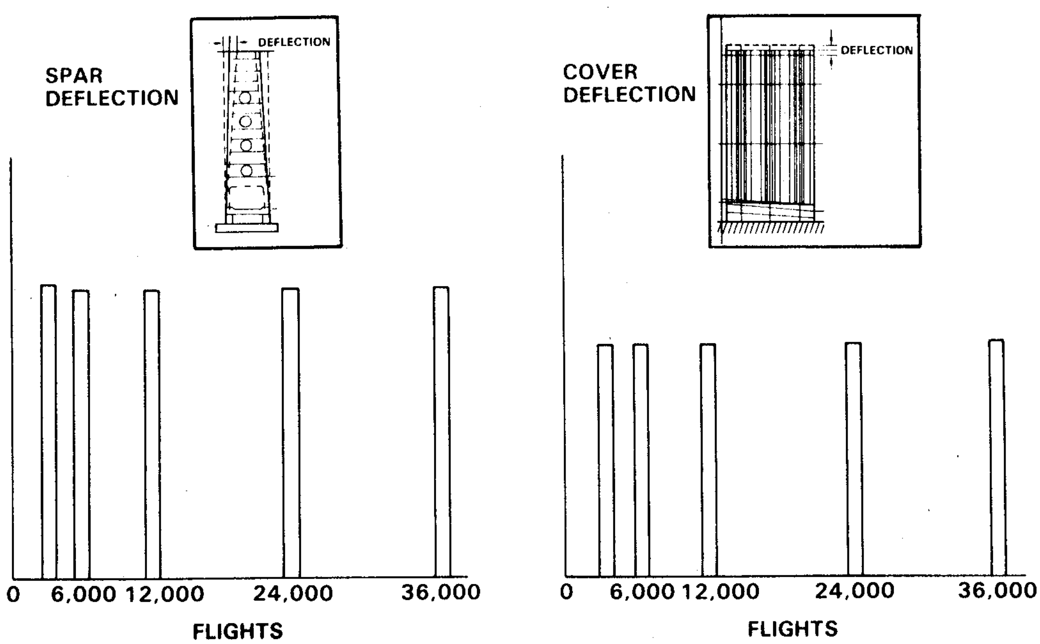


Figure 22

SUMMARY

The three questions posed at the beginning of the program have now been answered.

The range of production qualities that can be expected for components manufactured under conditions similar to those expected in production has been established. The spars were produced using tooling which underwent only minor modifications during the run of 24 components, similar to a production run. The covers were fabricated using tooling that underwent various modifications during a run of 28 components. Thus extremes of the production environment were encountered.

The quality control procedures used proved adequate in identifying discrepancies. In particular NDI techniques developed and refined during the program worked very well. The mechanical process control tests proved to be of varied effectiveness individually but when viewed on a combined basis for each component correlated well with NDI and physical tests.

Figure 23 depicts the main conclusions.

REPRODUCIBILITY WAS DEMONSTRATED

NO REJECTIONS DUE TO MATERIAL

NO REJECTIONS DUE TO PROCESSING

REJECTIONS DUE TO:

- **TOOLING PROBLEMS**
- **MACHINING ERRORS**
- **CARELESSNESS**

Figure 23

Figure 24 summarizes the major test conclusions.

The static test results showed excellent uniformity. The coefficients of variation (CV), 3.3 percent for the cover and 6.1 percent for the spars compare favorably with those of other common structural materials. The allowables used were derived from coupon data. The failure modes of the covers and spars are influenced primarily by stiffness. The specimens in all cases failed at loads higher than predicted. The allowables used for prediction were based on average coupon data whereas design allowables are statistically reduced below those levels. The allowable thus proved adequate to account for structure static strength variability.

The durability testing showed that the combined effects of long-term cyclic environment and cyclic loads below design limit load have no deleterious effects.

STATIC TESTS

- **EXCELLENT UNIFORMITY**
- **CV_S 3.3 TO 6.1% COMPARED TO 3 TO 10% FOR OTHER MATERIALS**
- **CONSISTENT FAILURE MODES**

DURABILITY TESTS

- **STIFFNESS NOT AFFECTED**
- **MOISTURE INGESTION BELOW DESIGN LEVEL OF 1%**
- **KNOWN DEFECTS DID NOT GROW AT DESIGN STRAIN LEVELS**
- **CORROSION PROTECTION BETWEEN ALUMINUM AND GR/EP WAS DEMONSTRATED**
- **REPAIRS WERE DEMONSTRATED**

Figure 24

COMPOSITE WING FUEL CONTAINMENT AND DAMAGE TOLERANCE—TECHNOLOGY DEVELOPMENT

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ACEE Composite Structures Technology Conference
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PROGRAM OBJECTIVE

In October 1981, the Lockheed-California Company began a two-phase program to identify and resolve technical problems associated with fuel containment and damage tolerance of composite material primary wing structure for transport aircraft. The program objective is defined in Figure 1. The program is sponsored by the National Aeronautics and Space Administration as part of the Aircraft Energy Efficiency (ACEE) Composites Structures Program. This paper presents the results of the technology development portion of this program.

IDENTIFY AND RESOLVE TECHNICAL ISSUES ASSOCIATED WITH FUEL CONTAINMENT AND DAMAGE TOLERANCE OF COMPOSITE WINGS FOR TRANSPORT AIRCRAFT

Figure 1

PROGRAM TASKS

The technology development phase of the program included the following activities: preliminary design of composite material wing surfaces for a transport aircraft, evaluation of high strain-to-failure graphite fiber/toughened resin composites, the investigation of lightning strike behavior of stiffened composite material panels, and the evaluation of fuel sealing methods for bolted joints. The second phase of the program will demonstrate the technology developed during the first phase through the fabrication and test of a large portion of a wing cover assembly. The program tasks are defined in Figure 2. Results from the second phase of the program will be presented in a subsequent paper.

- **PRELIMINARY DESIGN STUDIES OF COMPOSITE MATERIAL WING SURFACES**
- **INVESTIGATION OF TOUGHENED RESIN COMPOSITES WHICH OFFER IMPROVED DAMAGE TOLERANCE**
- **INVESTIGATION OF FUEL CONTAINMENT CONCEPTS FOR JOINTS AND IMPACT DAMAGE**
- **EVALUATION OF LIGHTNING STRIKE BEHAVIOR**
- **FABRICATION AND TEST OF TECHNOLOGY DEMOSTRATION ARTICLE**

Figure 2

WING COVER DESIGN STUDIES - DESIGN REQUIREMENTS

The baseline wing selected for the study was from the L-1011 transport aircraft. To guide the conceptual design activity a criteria document was compiled which included the wing geometry, loads, stiffness requirements, environmental conditions, and manufacturing constraints, Figure 3. Inplane loads and stiffnesses were used in conjunction with out-of-plane loads such as those due to fuel pressure to do the preliminary sizing of the wing surface structure. Environmental conditions considered for materials selection included: temperature extremes of -65°F to 180°F, resistance to fluids such as fuel, hydraulic fluid and water, and Zone 2 lightning strikes. The damage tolerance criteria stated that for cases where the damage cannot be detected by visual inspection, the structure shall be designed such that the damaged structure can withstand design ultimate loads. For large damage, such as might occur during flight due to an uncontained engine failure, the structure must be able to withstand design limit load.

- **BASELINE AIRCRAFT—L-1011**
- **END LOAD RANGE 2.3 TO 26 KIPS/IN**
- **SHEAR STIFFNESS RANGE 542 TO 932 KIPS/IN**
- **INTEGRAL FUEL TANKS**
- **ZONE 2 LIGHTNING STRIKE**
- **TEMPERATURE RANGE -65°F TO 180°F**
- **TWO SPAR WING**
- **26 IN. RIB SPACING**

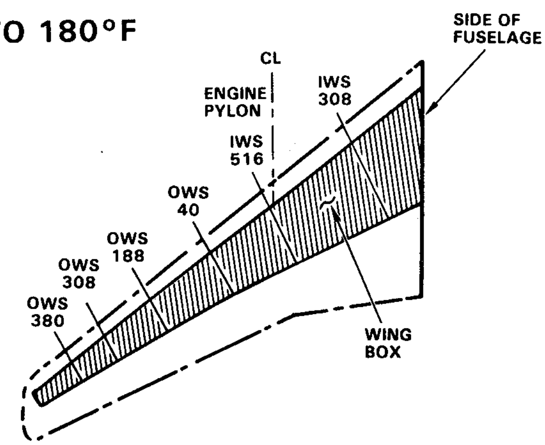
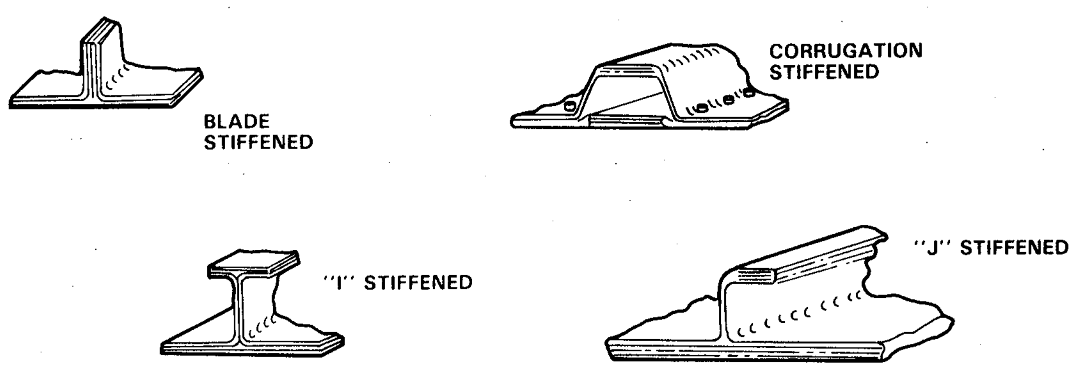


Figure 3

WING COVER DESIGN STUDIES - DESIGN CONCEPTS INVESTIGATED

Several designs, shown in Figure 4, were evaluated for the upper and lower wing covers. Each design was optimized relative to skin thickness and orientation, and stiffener geometry and spacing for the structural criteria and manufacturing constraints. For enhanced damage tolerance, stiffener geometries and/or skin laminate orientations were analyzed which offer resistance to delamination propagation.

Results of design trade-offs indicated very little difference in weight savings potential between the various concepts. Potential weight savings can be increased if the design strain levels of the materials can be improved from current values.



STUDY RESULTS

- PREDICTED WEIGHT SAVINGS 25% TO 30% USING APPLIED STRAIN CONSTRAINT OF 4000μ IN/IN
- WEIGHTS SAVINGS CAN BE INCREASED BY 12% IF STRAIN ALLOWABLE WERE 6000μ IN/IN

Figure 4

EVALUATION OF TOUGHENED RESIN COMPOSITES

Current applications of composite materials to aircraft structure, most of which are stiffness critical secondary structural components and medium size primary structural components, have demonstrated weight savings from 20 percent to 30 percent. The greatest impact on aircraft performance and cost will be made when these materials are used for fabrication of primary wing and fuselage structures which are 30 to 40 percent lighter than their metal counterparts.

High strain graphite fibers, in conjunction with toughened resins offer a potential to increase design allowable strain levels. As a part of this program, several materials were investigated by conducting processing studies and structural tests. Figure 5 lists the objective and tests conducted. The types of tests performed are described in NASA Reference Publication 1092, July 1983, "Standard Tests for Toughened Resin Composites." The test data included in this presentation includes the notched tensile strength, impact damage characteristics and post-impact compression strength.

OBJECTIVE:

- **COMPARE THE STRUCTURAL BEHAVIOR OF TOUGHENED AND UNTOUGHENED RESIN GRAPHITE/ EPOXY MATERIALS**

PRINCIPAL TESTS:

- **NOTCHED TENSILE STRENGTH**
- **IMPACT BEHAVIOR**
- **POST-IMPACT COMPRESSION STRENGTH**

Figure 5

EVALUATION OF TOUGHENED RESIN COMPOSITES - MATERIALS INVESTIGATED

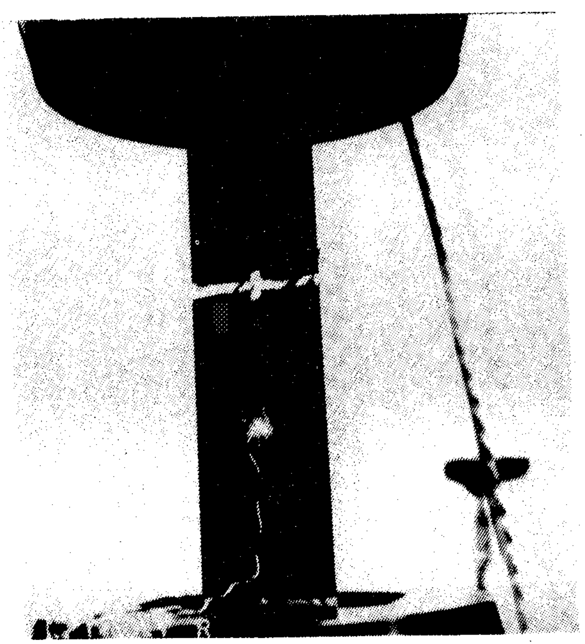
Four toughened resin composites and one untoughened (baseline) resin composite were evaluated, Figure 6. The baseline system was Hercules AS4/3502. The toughened systems included Hercules AS4/2220-1, American Cyanamid Celion/982, Narmco high strain Celion/5245C, and Hexcel high strain Celion/1504.

FIBER	RESIN	RESIN TYPE
AS4	3502	EPOXY
AS4	2220-1	TOUGHENED EPOXY
CELION	982	TOUGHENED EPOXY
HIGH STRAIN CELION	5245C	BISMALEIMIDE
HIGH STRAIN CELION	1504	TOUGHENED EPOXY

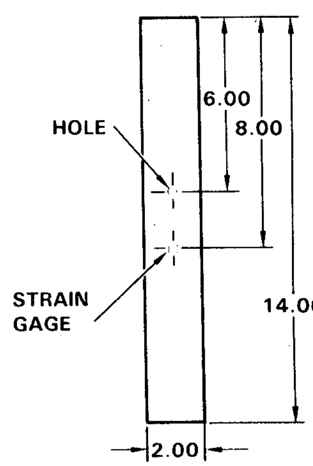
Figure 6

EVALUATION OF TOUGHENED RESIN COMPOSITES -
TENSION TEST FIXTURE AND SPECIMEN

The notched tensile strengths of the materials were determined by conducting tensile tests on coupons having open holes. Coupons 2.0 inches wide by 14.0 inches long with a 0.25 inch diameter hole were tested. Axial strain gages were used to determine the far-field strains within the coupon. The test setup is shown in Figure 7.



TEST FIXTURE



TEST SPECIMEN
(DIMENSIONS IN INCHES)

Figure 7

EVALUATION OF TOUGHENED RESIN COMPOSITES -
NOTCHED TENSION STRENGTH

A minimum of three tests were conducted for each material, laminate orientation, and test condition. Comparing the AS4/3502 data with the AS4/2220-1 data indicates that the tougher resin allows superior translation of fiber strength. The strain to failure of all the toughened resin materials far exceeded the current design allowable strain. A comparison of several materials is shown in Figure 8.

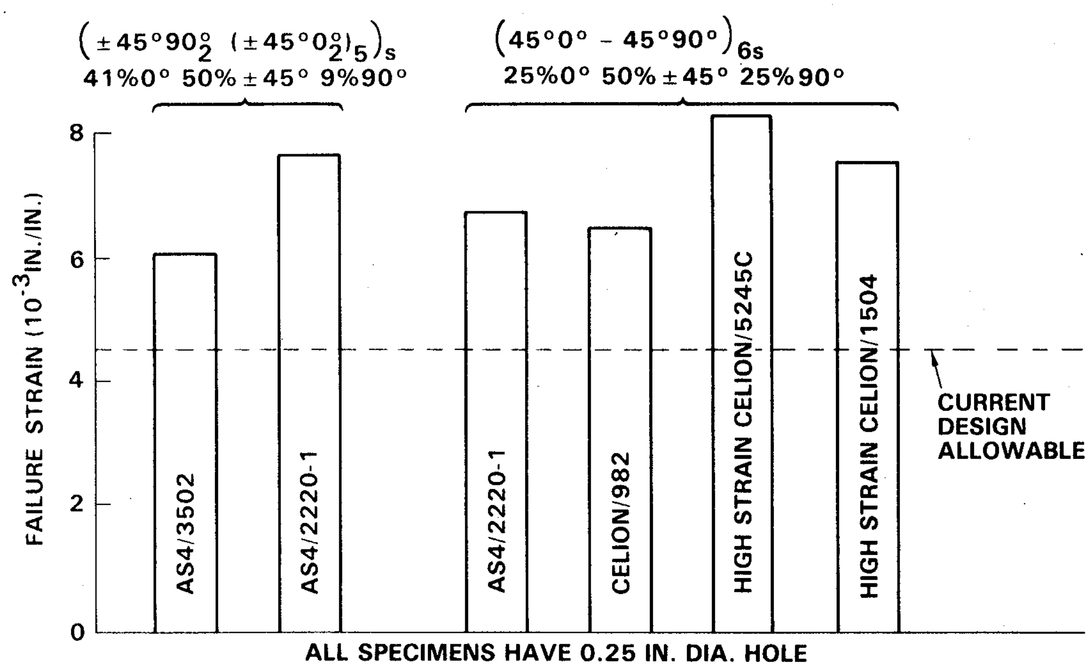


Figure 8

EVALUATION OF TOUGHENED RESIN COMPOSITES -
IMPACT TEST FIXTURE

Quasi-isotropic panels, 48 plies thick were fabricated with each material and subjected to impact tests. For these tests a 25 inch by 7 inch portion of the laminate was clamped to a steel plate with a 5 inch by 5 inch opening. The panel was struck in the center of the opening with a 12 pound impactor which had a 1/2 inch hemispherical diameter hardened steel tip. The impact test fixture is shown in Figure 9.

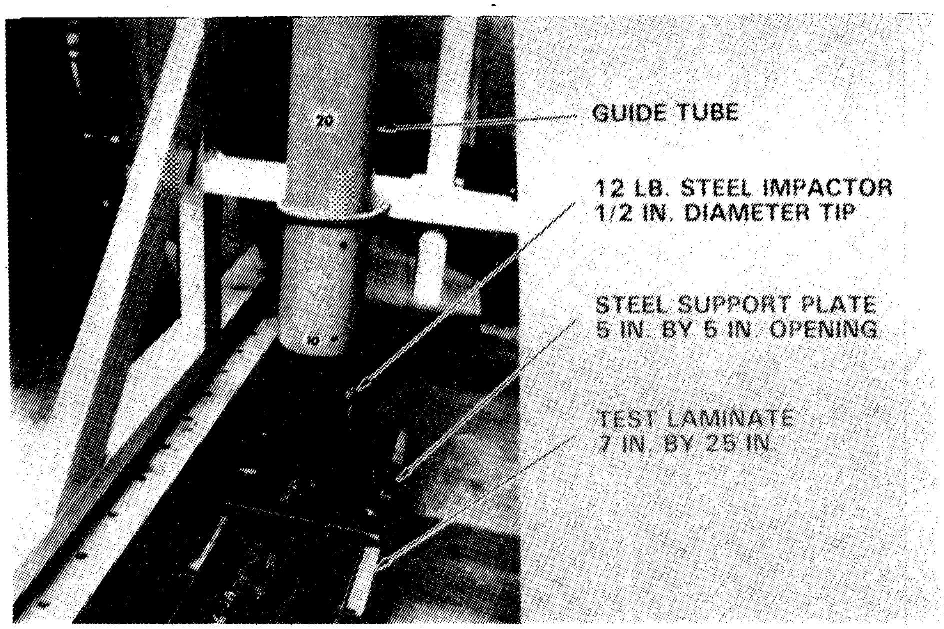


Figure 9

EVALUATION OF TOUGHENED RESIN COMPOSITES - IMPACT RESPONSE

After impacting, the panels were inspected visually and ultrasonically to ascertain the amount of damage. The results are shown in Figure 10. In general, the impact damage in the toughened resin composites was greater than that in the baseline system for a given impact energy. The impact energy to cause initial damage (ultrasonically detectable) and front surface visual damage was similar for each material. The impact energy to puncture the laminate was higher for the toughened resin systems than the baseline material.

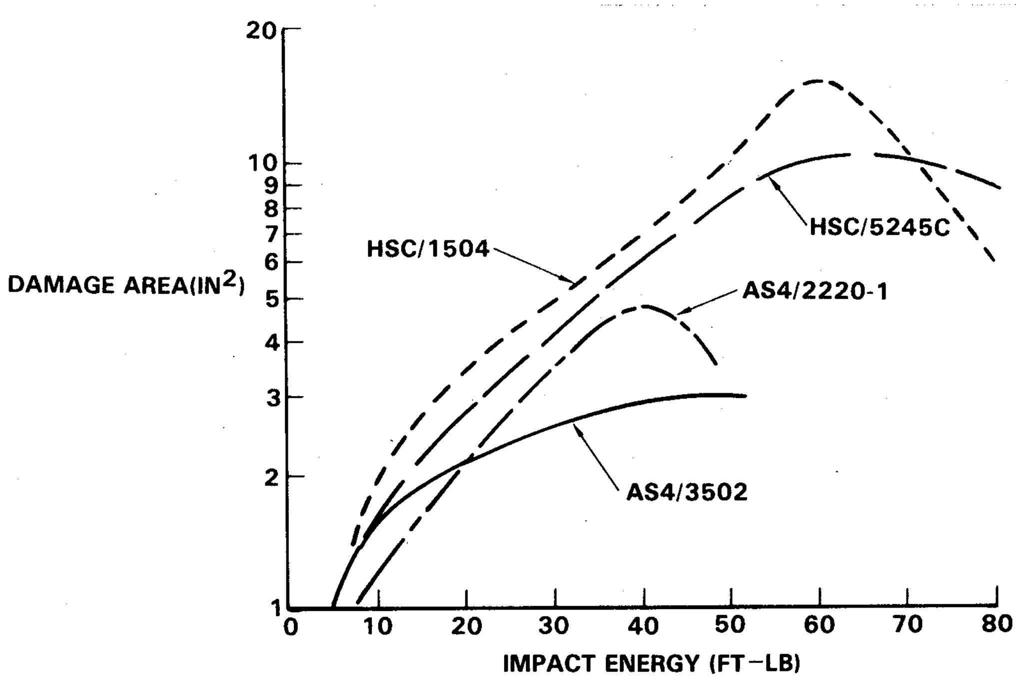


Figure 10

EVALUATION OF TOUGHENED RESIN COMPOSITES - COMPRESSION TEST FIXTURE AND SPECIMEN

Tests were conducted on each material to determine the effect of impact damage on compression strength. A 7 inch by 25 inch laminate was first impacted at two locations and then inspected for damage. Two 5 inch by 10 inch coupons were then machined from the laminate. The test fixture, Figure 11, simply-supported the coupon at the sides and clamped it at the loaded edges. This technique of stabilizing the coupon allows the out-of-plane deflections associated with delamination growth. Each coupon was instrumented with back-to-back axial strain gages located away from the damaged area.

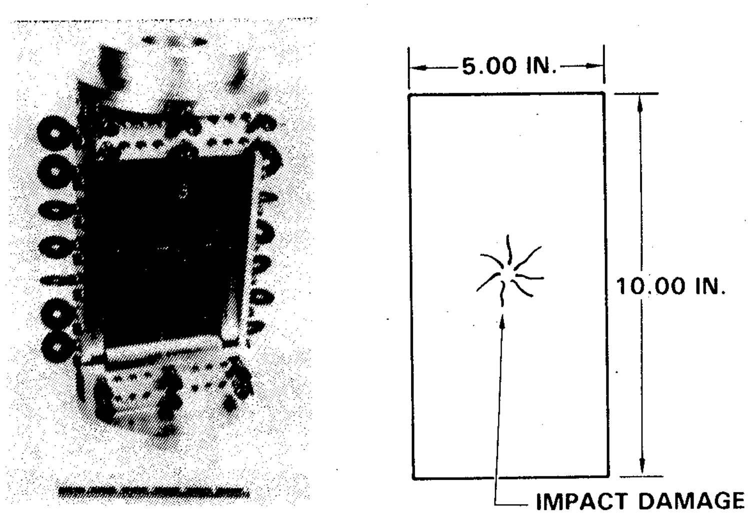


Figure 11

POST IMPACT COMPRESSION STRENGTH

Post-impact compression tests were conducted on each material for two impact energies, 20 ft-lb and 30 ft-lb. These energies were selected because the resulting damage is visible on the impacted surface of the laminate. For each material and laminate orientation tested, the compression failure strain is plotted versus the damage area measured from the ultrasonic inspections. The data points shown in Figure 12 represent a minimum of three tests.

For the tests conducted at an energy level of 20 ft-lb, the failure strain of the toughened systems was, in some cases, much better than the baseline material. However, at the 30 ft-lb energy level, the improvement in failure strain of the toughened systems over the baseline material was minimal. This can be attributed to the greater amount of damage in the toughened systems compared to the baseline material. Based on this data it would appear that none of the toughened materials offers the improvement in impacted compression strain-to-failure needed to substantially increase design allowable compression strains.

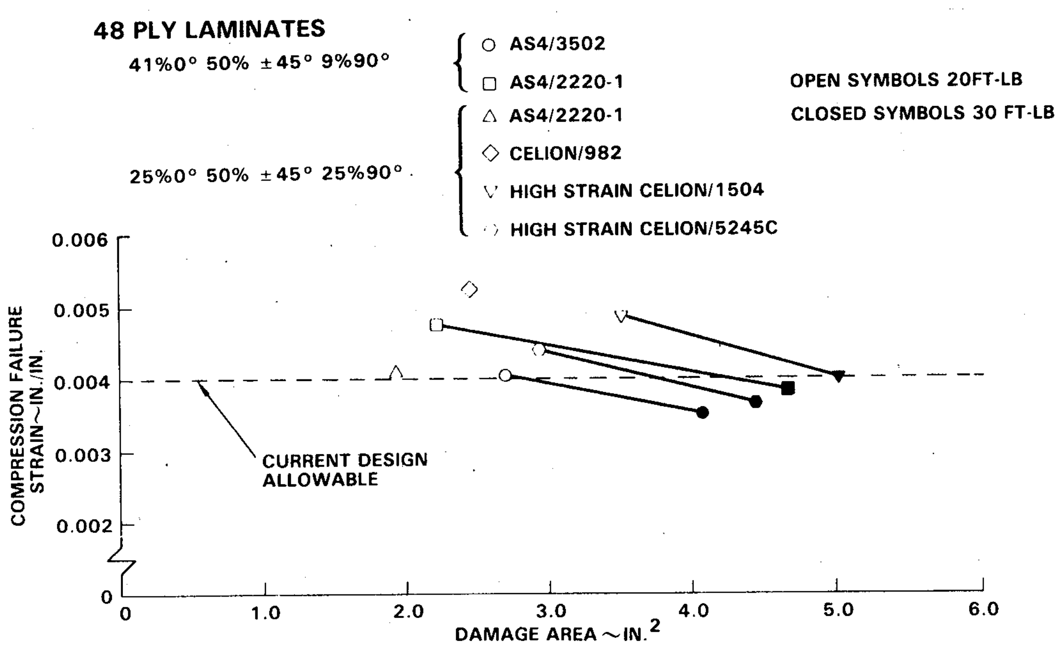


Figure 12

FUEL CONTAINMENT - EVALUATION OF SEALING MATERIALS

Cover-to-substructure and cover spanwise joints are potential sources of leakage in a wing box which contains fuel. Two methods were evaluated to seal mechanically fastened joints: the conventional approach using a sealant, and an adhesively sealed joint. Comparative tests were conducted using the single lap specimen shown in Figure 13. The specimen was designed to be critical in bolt bearing and had a design ultimate load of 6460 lb. All coupons were constructed of AS4/3502. One-half of the specimens were sealed with a polysulfide sealant and the remainder were bolt-bonded with an AF-10 adhesive. Fillets and fastener collars were sealed with polysulfide sealant on both types of coupons. The fuel simulant used for the tests was Shell Pella A with fluorescent dyes added to enhance visibility with ultraviolet light.

OBJECTIVE:

- **EVALUATE TECHNIQUES TO SEAL BOLTED JOINTS**

SCOPE:

- **LAP SHEAR SPECIMEN USED FOR COMPARATIVE TESTS**
- **INVESTIGATED POLYSULFIDE SEALANT AND AF-10 ADHESIVE**

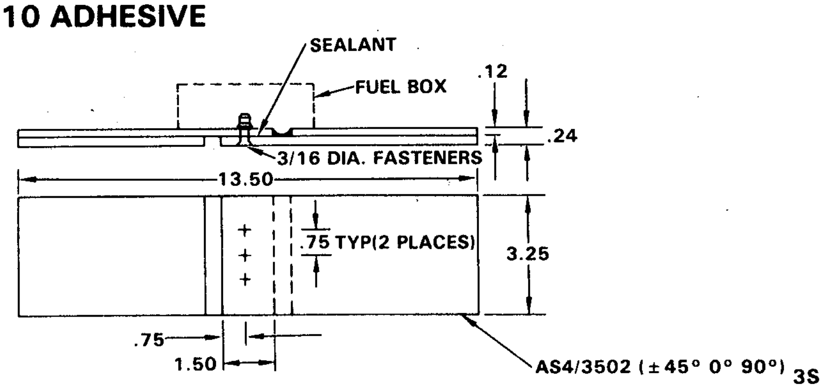


Figure 13

FUEL CONTAINMENT - EVALUATION OF SEALING MATERIALS

Three types of tests were conducted on each group of specimens; sustained load, cyclic-load, and cyclic temperature. Fuel pressure was applied to all specimens during the tests. None of the specimens leaked during sustain load, cyclic load, or thermal cycles. Static strength tests were conducted on the sustained load and thermally cycled specimens. No specimen leaked prior to rupture, and the failure load of all specimens exceeded design ultimate load. The results of the tests are shown in Figure 14.

TEST CONDITION	TEST RESULTS	
	POLYSULFIDE	AF-10
• 300 HRS AT 1930 LB TENSION AND 15 PSI	NO LEAKS	NO LEAKS
• 200 HRS AT 4000 LB TENSION AND 15 PSI	NO LEAKS	NO LEAKS
• 15 MIN. AT 4000 LB TENSION AND 20 PSI	NO LEAKS	NO LEAKS
• RESIDUAL STRENGTH TEST AT 15 PSI	6590 LB (1)	10080 LB (1)
• 36,000 CYCLES: 1930 LB, −579 LB AND 15 PSI	NO LEAKS	NO LEAKS
• 36 CYCLES: 3088 LB, −926 LB AND 15 PSI	NO LEAKS	NO LEAKS
• 100 CYCLES: −65°F TO 140°F AND 15 PSI	NO LEAKS	NO LEAKS
• RESIDUAL STRENGTH TEST AT −65°F AND 15 PSI	7200 LB (1)	8690 LB (1)

(1) AVERAGE OF 2 SPECIMENS, DESIGN ULT LOAD = 6460 LB.

Figure 14

Based on the results of lap shear tests, it was concluded that the joints sealed with polysulfide met all design requirements and would be less costly to assemble than adhesively bonded joints. To verify the applicability of polysulfide sealant for complex composite structures, three box beams were designed and fabricated. Two fastener types and spacings were evaluated with the three beam specimens.

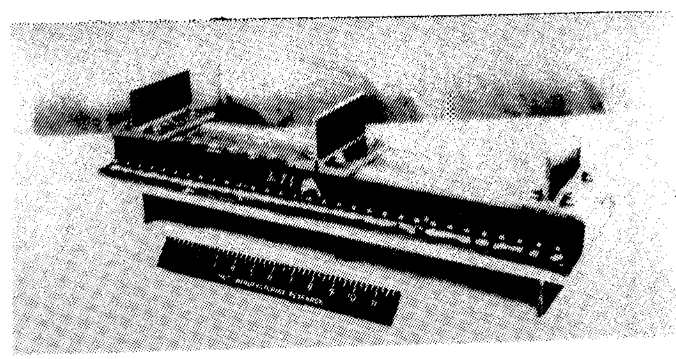
The configuration of the beams, shown in Figure 15, was chosen to simulate fuel leak paths and loading conditions typical of a spar cap to cover joint.

OBJECTIVE:

- **VERIFY APPLICABILITY OF POLYSULFIDE SEALANT FOR COMPOSITE STRUCTURES**

SCOPE:

- **BOX BEAM TESTS INVESTIGATED TWO FASTENER TYPES AND SPACINGS**



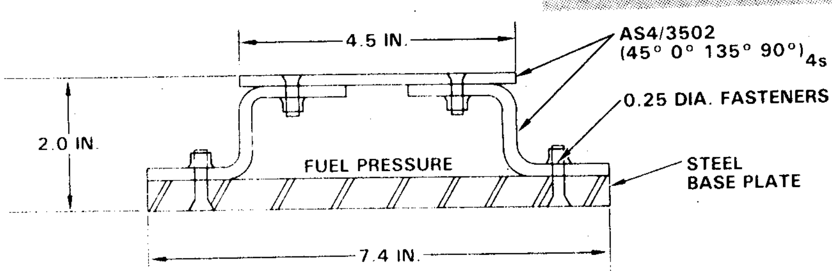


Figure 15

FUEL CONTAINMENT - FUEL SEALING VERIFICATION

The three specimens were leak tested at 6 psi, fatigue tested for 36,000 cycles at 50 percent limit load (R = -0.5) and 36 cycles at 80 percent limit load (R = 0.5) with 6 psi fuel pressure, and then residual strength tested in combination with 15 psi fuel pressure. None of the specimens leaked during the fatigue tests. The specimens did not leak during the residual strength tests until the graphite/epoxy cover laminate ruptured in tension. The results of the test are shown in Figure 16.

BEAM I.D.	FASTENER TYPE AND SPACING	TEST RESULTS		
		FATIGUE ①	RESIDUAL ② STRENGTH ③ (LB.)	FAILURE STRAIN (IN/IN)
1	TRIWING SCREWS HI-LOK COLLARS-4.5D	NO LEAKS	23,850	0.0047
2	TRIWING SCREWS HI-LOK COLLARS-6.0D	NO LEAKS	20,000	0.0044
3	HUCK GROOVE PROPORTIONED LOCKBOLTS-4.5D	NO LEAKS	23,510	0.0047

① 36,000 CYCLES +6000 LB/-3000 LB AND 6 PSI
36 CYCLES +9600 LB/-4800 LB AND 6 PSI

② COMBINED WITH 15 PSI

③ DESIGN ULTIMATE LOAD = 18.0 KIPS.

Figure 16

FUEL CONTAINMENT - POST-IMPACT FUEL LEAKAGE

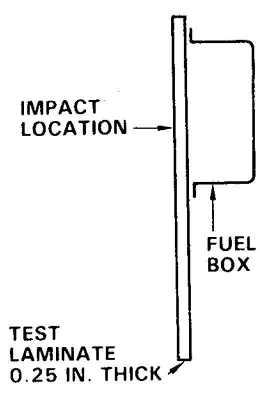
Another potential source of fuel leakage in a wing box constructed with graphite/epoxy is impact damage. A preliminary evaluation of this threat was made using 0.25 inch thick unpainted graphite/epoxy panels impacted at various energy levels and then subjected to fuel pressure on the side opposite to the impact. As shown in Figure 17, impacted samples of AS4/2220-1 and AS4/3502 leaked fuel after a very short time at low fuel pressure. In fact, one specimen, impacted at 15 ft-lb leaked within 72 hours with just the full fuel box (approximately 2 inches deep) placed on top of the specimen.

OBJECTIVE:

- **EVALUATE POST-IMPACT FUEL LEAKAGE OF GRAPHITE/EPOXY LAMINATES**

SCOPE:

- **CONDUCT IMPACT AND LEAK TESTS ON COATED AND UNCOATED LAMINATES**



UNCOATED LAMINATES	IMPACT ① ENERGY (FT-LB)	FUEL PRESSURE (PSI)	TIME TO LEAK (HRS)
AS4/3502	10	10	1.0
AS4/3502	10	5	1.0
AS4/3502	15	0	< 72
AS4/2220	10	10	3.3
AS4/2220	20	5	< 22

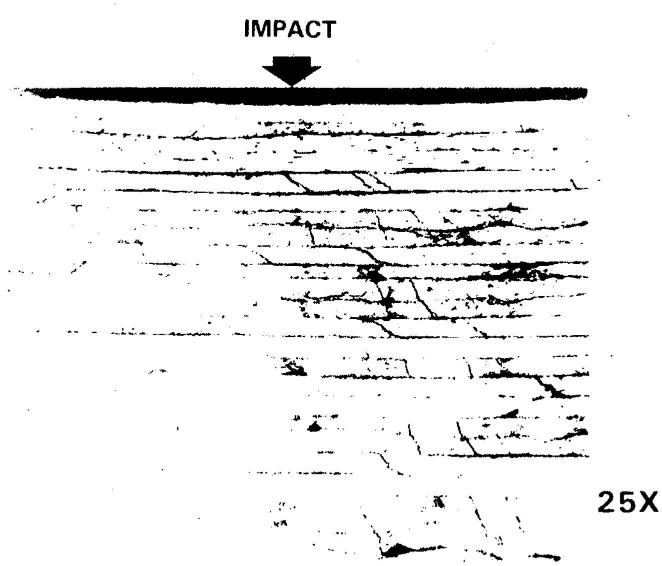
① NO PRESSURE APPLIED AT IMPACT

Figure 17

FUEL CONTAINMENT - POST-IMPACT FUEL LEAKAGE

This photomicrograph, Figure 18, shows the internal damage to a 0.25 inch thick AS4/3502 laminate caused by a 10 ft-lb impact. Note, that at these low impact levels the impact damage was not visually detectable and that neither the front surface nor the back surface of the laminate appeared to be ruptured. Yet this laminate leaked fuel within 1.0 hour at a pressure of 10 psi.

INTERIOR IMPACT DAMAGE



10 FT-LB IMPACT DAMAGE ON A 48-PLY AS4/3502 LAMINATE

Figure 18

FUEL CONTAINMENT - POST-IMPACT
FUEL LEAKAGE TEST LAMINATES

Two techniques, exterior coating and an embedded plastic film, were investigated to determine the lightest weight method to prevent fuel leaks for non-visible impact damage to graphite/epoxy laminates. The test panels, shown in Figure 19, were 32-ply quasi-isotropic laminates constructed with AS4/2220-1. All panels were painted on the impacted surface with an epoxy primer and polyurethane topcoat.

The panels were impacted at energies of 10 ft-lb, 20 ft-lb, and 30 ft-lb using the impact test fixture described previously. After nondestructive inspection, the panels were subjected to fuel leak tests.

- **SEVERAL COATINGS WERE EVALUATED TO PREVENT FUEL LEAKS FOR NON-VISIBLE IMPACT DAMAGE**

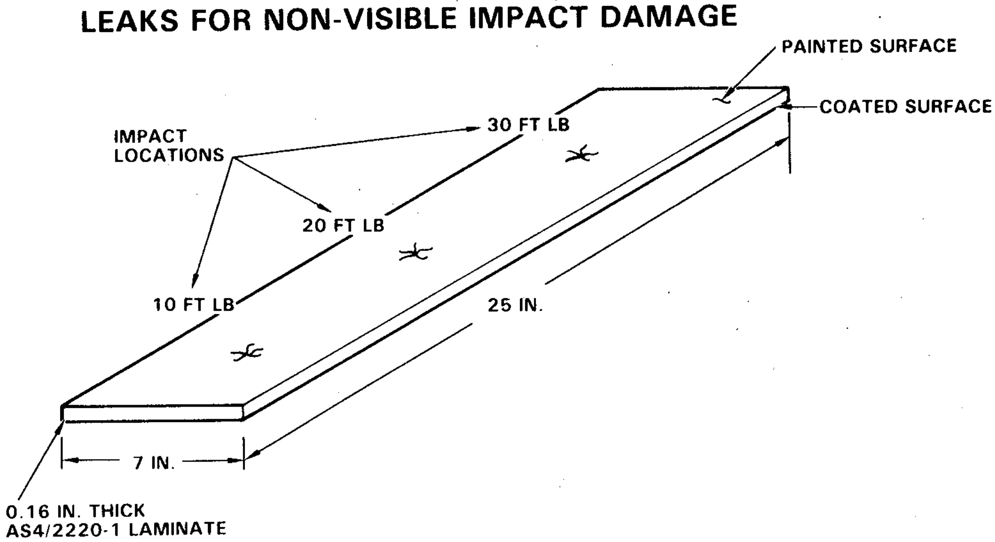


Figure 19

FUEL CONTAINMENT - POST-IMPACT FUEL LEAKAGE

Five panels were tested. The first had no treatment and was used as a control. The second had a 0.013 inch thick polyurethane film laminated at the midplane of the panel. The third, fourth, and fifth panels were coated on the interior surface of the laminate (side opposite the impact) with a polyurethane coating called Chemglaze. Two thicknesses were evaluated, 5 mil and 10 mil.

Results of the post-impact fuel leak tests, shown in Figure 20, indicate that the 5 mil coating of Chemglaze was the most efficient method to eliminate fuel leakage for low energy impacts.

CONFIGURATION	COATING WEIGHT (LB/FT^2)	TIME TO LEAK		
		10 FT-LB	20 FT-LB	30 FT-LB
UNCOATED	—	NO LEAK ①	14 MIN., 0 PSI	1 SEC., 0 PSI
POLYURETHANE FILM (0.013 IN.) AT MIDPLANE	0.080	NO LEAK ①	NO LEAK ①	16 MIN., 0 PSI
CHEMGLAZE ② (0.005 IN.)	0.049	NO LEAK ①	NO LEAK ①	NO LEAK ①
CHEMGLAZE ② (0.010 IN.)	0.098	NO LEAK ①	NO LEAK ①	NO LEAK ①
CHEMGLAZE ② (0.005 IN.) AND FIBERGLASS FABRIC (0.005 IN.)	0.087	NO LEAK ①	NO LEAK ①	6 MIN., 9 PSI

① 50 HR AT 10 PSI

② LORD CHEMICAL PRODUCTS

Figure 20

LIGHTNING STRIKE BEHAVIOR

A potential problem with fuel containing wing boxes constructed with graphite/epoxy is fuel ignition due to a lightning strike. The majority of the wing box surface is classified as Zone 3 (current transfer region); however, the area behind the engine is a Zone 2 (swept-stroke) region.

To evaluate the lightning strike behavior of graphite/epoxy wing skins, stiffened panels were fabricated and tested, Figure 21. A variety of surface protection materials and fastener treatments were investigated to determine the lightest weight technique to eliminate arcing and minimize structural damage.

REQUIREMENTS
- ZONE 2, SWEPT STROKE
- NO ARCING IN FUEL TANK
- MINIMAL STRUCTURAL DAMAGE

VARIABLES INVESTIGATED
- SURFACE PROTECTION MATERIALS
- FASTENER TREATMENTS

TEST SPECIMEN GEOMETRY

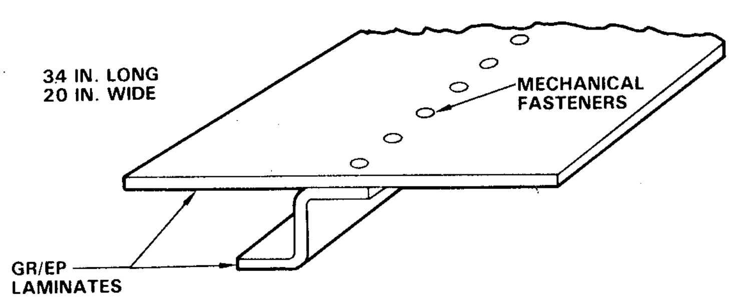


Figure 21

LIGHTNING STRIKE BEHAVIOR -
TEST ARRANGEMENT FOR SWEPT-STROKE TESTS

All of the panels were tested by Lightning and Transients Research Institute for 100,000 ampere swept-stroke lightning current levels. The test setup is depicted in Figure 22. A camera was used to determine if sparking occurred during the test. Upon the completion of the tests, the panels were inspected visually and ultrasonically to determine the amount of damage.

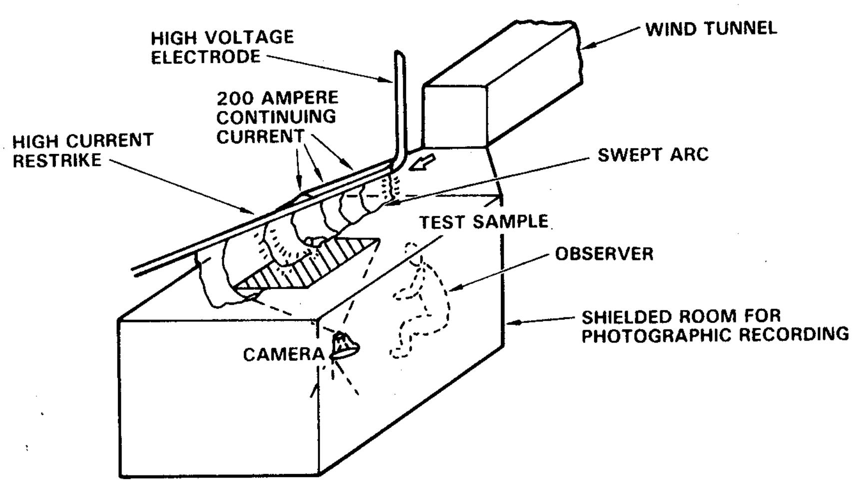
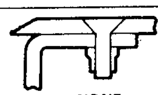
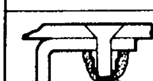
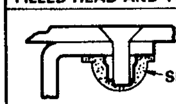


Figure 22

LIGHTNING STRIKE BEHAVIOR - FASTENER TREATMENTS AND SURFACE PROTECTION

Swept-stroke lightning tests were conducted on graphite/epoxy panels which had a graphite/epoxy stiffened bolted to it to simulate a rib to cover joint. Three types of surface protection materials and three fastener treatments were evaluated in comparison to panels with no surface protection or fastener treatments. Test results, Figure 23, indicate that a surface protection of graphite fabric with 8 mil aluminum wires located on 1/8 inch centers in combination with a fastener treatment of polysulfide topcoat with a plastic cap would minimize structural damage and eliminate arcing.

FASTENER TREATMENT	SURFACE PROTECTION		NUMBER OF TESTS	NUMBER OF TESTS WHICH CAUSED ARCS
	DESCRIPTION	WEIGHT (LB/FT²)		
 NONE	8 MIL AL WIRE/ GRAPHITE FABRIC ①	0.0128	3	1
 SEALANT POLYSULFIDE TOPCOAT	NONE	--	6	6
	8 MIL AL WIRE/ GRAPHITE FABRIC ①	0.0128	5	1
SEALANT  SEALANT RECESSED AND FILLED HEAD AND TOPCOAT	NONE	–	3	3
	NICKEL PLATED GRAPHITE FABRIC	0.0410	4	1
	4 MIL AL WIRE/ GRAPHITE FABRIC ①	0.0028	2	2
	8 MIL AL WIRE/ GRAPHITE FABRIC ①	0.0128	4	1
 SEALANT PLASTIC CAP	8 MIL AL WIRE/ GRAPHITE FABRIC ①	0.0128	6	0

① WIRES ON 1/8 IN. CENTERS, BOTH WARP AND FILL DIRECTIONS

Figure 23

LIGHTNING STRIKE BEHAVIOR - SURFACE PROTECTION

The use of a surface protection material consisting of a hybrid fabric of graphite yarn and 8 mil diameter aluminum wires on 1/8 inch centers dramatically reduced the amount of structural damage caused by swept-stroke lightning. Figure 24 shows the protected and nonprotected samples struck by lightning.

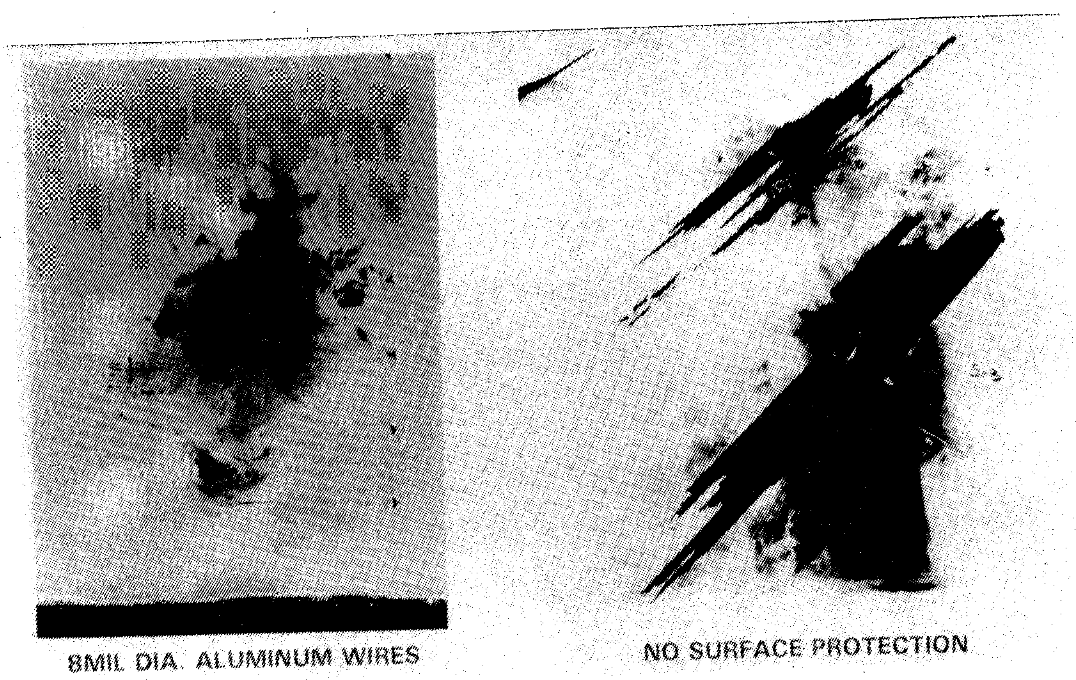


Figure 24

SUMMARY

Preliminary design studies predict that compared to the aluminum baseline, wing surfaces constructed with graphite/epoxy composites offer a large weight savings if design allowable strains can be increased from the current levels. Tests on laminates fabricated with high strain-to-failure graphite fibers combined with currently available tougher resins indicate that the desired strain allowable for tension can be obtained. However, for greater post-impact compression strength significant improvements are required.

Based on tests conducted in this program, it is concluded that the conventional fuel tank sealing techniques used for joints in metal structures are equally applicable to composite structures. However, the fuel containment capability of a graphite/epoxy tank could be compromised by low energy impact damage. It has been determined that a 0.005 inch thick coating of a flexible polyurethane paint on the inside of the wing skin would prevent fuel leaks due to low energy impact damage.

Swept-stroke lightning strikes to unprotected graphite/epoxy stiffened panels caused internal sparking and a large amount of structural damage. A surface protection material consisting of a graphite/aluminum wire fabric and a fastener treatment of polysulfide topcoat and a plastic cap proved effective in eliminating arcing and reducing structural damage.

- HIGH STRAIN FIBERS AND TOUGHER RESINS PERMIT GREATER TENSION DESIGN ALLOWABLES
- FIBER/RESIN IMPROVEMENTS REQUIRED TO ATTAIN GREATER POST-IMPACT COMPRESSION STRENGTH
- CONVENTIONAL FUEL TANK SEALING TECHNIQUES FOR JOINTS ARE APPLICABLE TO COMPOSITES
- INTERIOR SURFACES OF THE WING MUST BE COATED TO PREVENT POST-IMPACT FUEL LEAKAGE
- SURFACE PROTECTION AND FASTENER TREATMENTS SUPPRESS INTERIOR ARCING DUE TO SWEPT-STROKE LIGHTNING STRIKES

Figure 25

COMPOSITE WING FUEL CONTAINMENT AND DAMAGE TOLERANCE - TECHNOLOGY DEMONSTRATION

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Burbank, California

ACEE Composite Structures Technology Conference
Third Special Oral Review
August 13-16, 1984

TECHNOLOGY DEMONSTRATION OBJECTIVE

In October 1981, the Lockheed-California Company began a two-phase program to identify and resolve technical problems associated with fuel containment and damage tolerance of composite primary wing structure for transport aircraft. The program is sponsored by the National Aeronautics and Space Administration as part of the Aircraft Energy Efficiency (ACEE) Composites Structures Program. This paper presents the results of the technology demonstration portion of this program.

DEMONSTRATE THE TECHNOLOGY DEVELOPED FOR FUEL CONTAINMENT, LIGHTNING STRIKE PROTECTION AND DAMAGE TOLERANCE OF COMPOSITE WINGS FOR TRANSPORT AIRCRAFT

Figure 1

WING SURFACE DESIGN REQUIREMENTS

The stiffened panel for the technology demonstration article was sized for the Lockheed L-1011, outer wing station (OWS) 188, upper surface design requirements. This location on the wing is outboard of the wing engine pylon. The area has integral fuel tanks, and a Zone 2 lightning strike requirement. The baseline wing rib spacing of 26 inches was used to size the graphite/epoxy stiffened panel design. The design loads at OWS 188 consisted of an axial compression load of -12,972 lb/in, a shear load of 1804 lb/in and an outward burst pressure of 11.46 psi. The shear stiffness requirement was 858,000 lb/in.

- **BASELINE AIRCRAFT — L-1011**
- **INTEGRAL FUEL TANKS**
- **ZONE 2 LIGHTNING STRIKE**
- **26 INCH RIB SPACING**
- **LOADS AND STIFFNESS REQUIREMENT**

WING STATION	AXIAL LOAD (LB/IN)	SHEAR LOAD (LB/IN)	PRESSURE (PSI)	SHEAR STIFFNESS Gt (LB/IN)
OWS 188	−12,972	1804	11.46	858,000

Figure 2

GRAPHITE/EPOXY MATERIAL PROPERTIES

At the time that the material for the stiffened panel was chosen, the Phase 1 material screening tests had been completed. These tests had shown a slight advantage in mechanical properties and a significant improvement in processibility of the Hercules AS4/2220-1 material over the baseline Hercules AS4/3502 material. Based on these results, the Hercules AS4/2220-1 material was chosen for the fabrication of the stiffened panel. Lamina properties were used as input into the Lockheed COMAIN laminate analysis program to obtain laminate properties for detail design and stress analysis. Allowable strength of the AS4/2220-1 material in tension and compression was determined by taking 80 percent of the average strength values from the room temperature, dry, 0.25 inch diameter notched tension tests, and 20 ft-lb, impacted compression tests, as shown below in Figure 3.

- **HERCULES AS4/2220-1**
- **LAMINA PROPERTIES USED FOR DESIGN**

E_{11} (MSI)	E_{22} (MSI)	μ_{12}	G_{12} (MSI)
20.24	1.49	0.3102	0.70

- **48-PLY $[\pm 45/90_2(\pm 45/0_2)_5]_S$ LAMINATE PROPERTIES**

TEST AND CONDITION	FAILURE STRESS (KSI)	FAILURE STRAIN (μIN/IN)	MODULUS (MSI)
TENSION 0.25 IN. DIA HOLE 75°F, DRY	59.97	7670	9.48
COMPRESSION 20 FT-LB IMPACT 75°F, DRY	−39.95	−4751	8.65

Figure 3

STIFFENED PANEL CONFIGURATION

The stiffened panel for the technology demonstration article was 54 inches long by 18 inches wide with two integral blade stiffeners spaced six inches apart. The stiffeners were precured and machined before assembly into the panel. At the outer surface of the panel, one 0.010 inch thick ply of 8 mil diameter aluminum wire/graphite/epoxy prepreg fabric was cocured to the 21% 0°, 71% ±45°, 8% 90°, outer skin laminate. The stiffener insert was adhesively bonded to the outer skin and the inner skin. The outer and inner skins were cocured together. After the panel was cured, the top edges of each stiffener were machined to a 0.12 inch radius and two plies of 120 style fiberglass fabric were wet laminated over each stiffener.

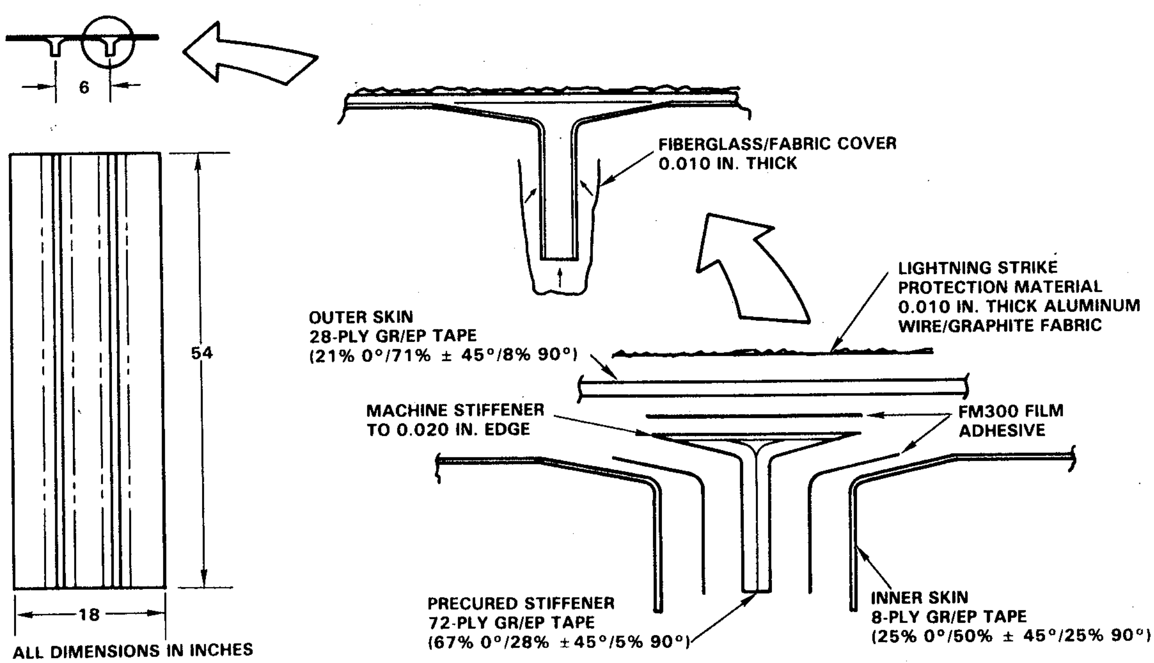
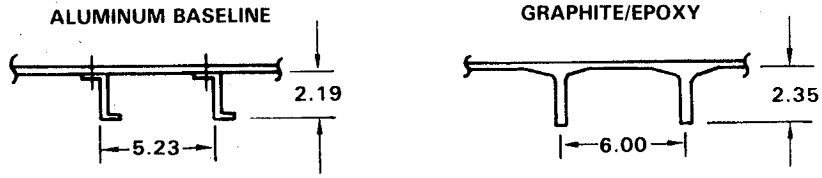


Figure 4

STIFFENED PANEL DESIGN SUMMARY

The aluminum baseline design at OWS 188 consisted of discrete 2.19 inch high, 'Z' stiffeners mechanically fastened to the skin, 5.23 inches apart. The graphite/ epoxy design was integrally stiffened with 2.35 inch high stiffeners, 6.00 inches apart. The axial stiffness of the graphite/epoxy design was 39 percent greater than that of the baseline design due to the lower, 4000μ in/in, design allowable compression strain of the graphite/epoxy material. Load sharing between the skin and the stiffeners of the graphite/epoxy design was 30 percent in the skin and 70 percent in the stiffeners as compared to 70 percent in the skin and 30 percent in the stiffeners in the baseline design. Skin shear stiffness of the graphite/ epoxy design was within 2 percent of the design requirement. Including the weight of the lightning strike protection and the Chemglaze fuel tank interior coating, the graphite/epoxy design weighed 20 percent less than the baseline aluminum design, as shown in Figure 5.

UPPER SURFACE CONFIGURATION



MATERIAL	ALUMINUM BASELINE	GRAPHITE/EPOXY
AXIAL STIFFNESS – $E\bar{t}$ (LB/IN)	3,180,000	4,422,000
% $E\bar{t}_{SKIN}$/% $E\bar{t}_{STIFFENER}$	70%/30%	30%/70%
SHEAR STIFFNESS – Gt (LB/IN)	880,000	843,000
DESIGN STRAIN (μ IN/IN)	6000	4000
AREAL WEIGHT (LB/IN^2)	0.0331	0.0266

20% WEIGHT SAVINGS

Figure 5

STIFFENER MANUFACTURING PROCESS DEVELOPMENT

Each stiffener was fabricated from two, high percent 0°, angle shaped laminates, a fillet made of 0° twisted tape and a 4-ply base laminate. These parts were assembled into a fixture for cure at 350°F. Following cure, the flanges of each stiffener were machined to a 7° taper in preparation for assembly into the stiffened panel assembly and curing fixture, shown below in Figure 6.

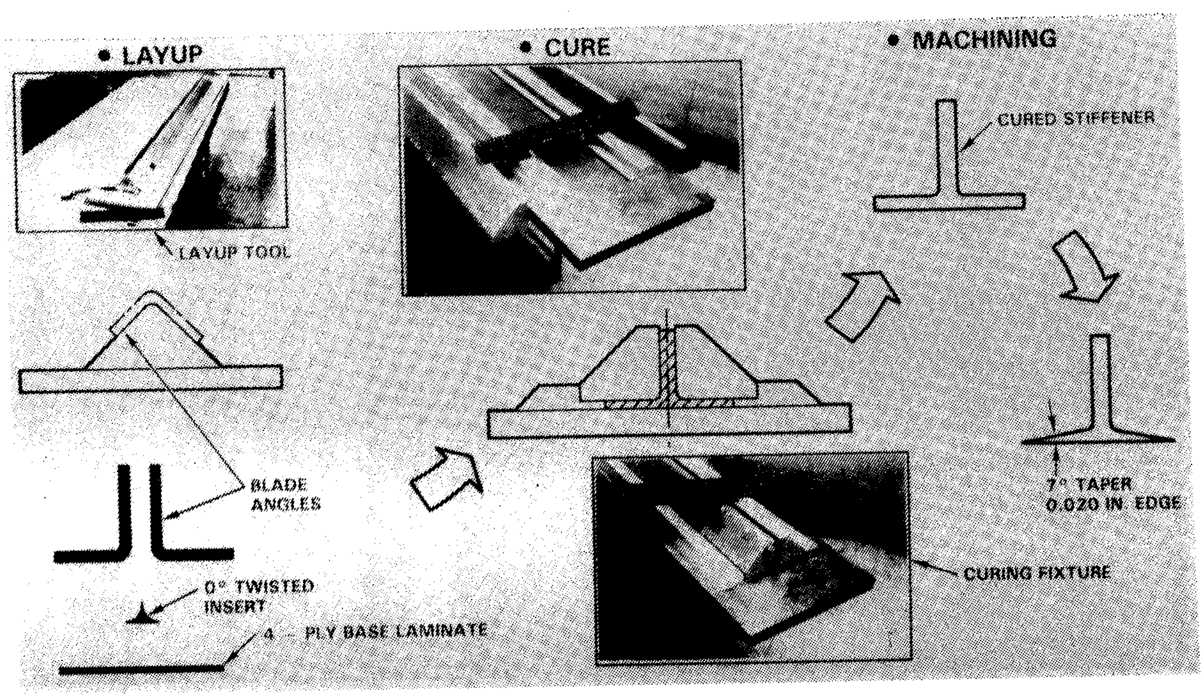


Figure 6

STIFFENED PANEL MANUFACTURING PROCESS DEVELOPMENT

In preparation for assembly and cure of the stiffened panel, the flat outer skin and the tooled inner skin sections of the panel were laid up. The component parts of the panel were put into the assembly and curing fixture starting with the inner skin sections, followed by the film adhesive covered precured stiffeners, the outer skin, and finally the lightning strike protection fabric. The stiffened panel assembly tool was fabricated from aluminum and silastic "J" rubber. The thermal expansion of the rubber and autoclave pressure on the 0.060 inch thick caul plate provided the pressure required during cure at 350°F. Due to the low viscosity of the AS4/2220-1 material during cure, few plies of bleeder material were required. Two process development stiffened panels and one stiffened panel for the technology demonstration article were fabricated.

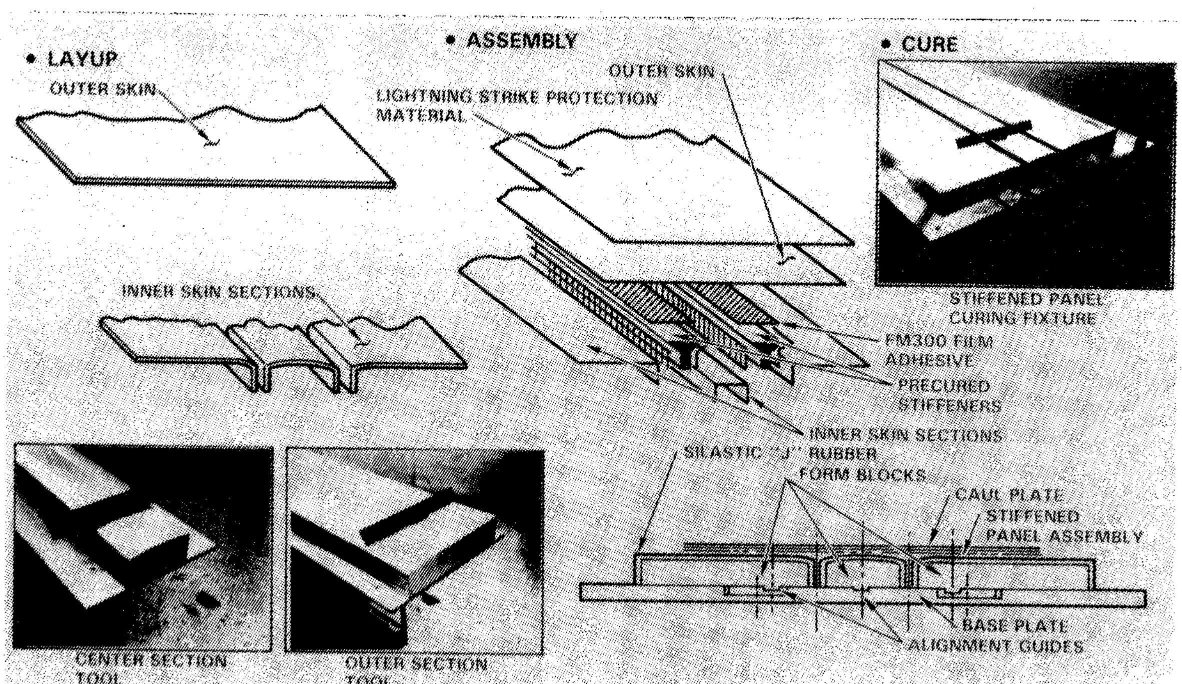


Figure 7

DESIGN DEVELOPMENT TEST OBJECTIVES

The objective of the design development tests was to verify that the stiffened panel structural details met the design requirements. Each stiffener was designed to withstand a pull-off load, induced at a stiffener-to-rib attachment by burst pressure in the wing, or side load, caused by a wing surface assembly mechanic standing on the side of a stiffener, of 300 pounds, distributed over three inches. The inner skins were designed to shear out the load from a broken stiffener in a length equal to three times the stiffener spacing, or 18 inches. The assembled panel was designed to withstand ultimate load with barely visible impact damage.

- **VERIFY THE INTEGRITY OF THE TECHNOLOGY DEMONSTRATION ARTICLE STIFFENED PANEL STRUCTURAL DETAILS**
  - **STIFFENERS DESIGNED FOR 100 LB/IN OVER 3 INCHES PULL-OFF LOAD AND SIDE LOAD**
  - **INNER SKIN DESIGNED TO SHEAR OUT THE LOAD FROM A BROKEN STIFFENER TO THE SURROUNDING PANEL IN THE LENGTH OF THREE (3) TIMES THE STIFFENER SPACING**
  - **STIFFENER AND SKIN DESIGNED TO TAKE ULTIMATE LOAD WITH BARELY VISIBLE IMPACT DAMAGE**

Figure 8

DESIGN DEVELOPMENT TEST PLAN

The design development tests were conducted on specimens cut from the two process development stiffened panels. Four stiffener pull-off load specimens and four stiffener side load specimens were tested. One stiffener failsafe specimen was tested in rail shear. One undamaged stiffener specimen and one impacted stiffener specimen were tested in compression. One two stiffener panel was impacted and tested in compression.

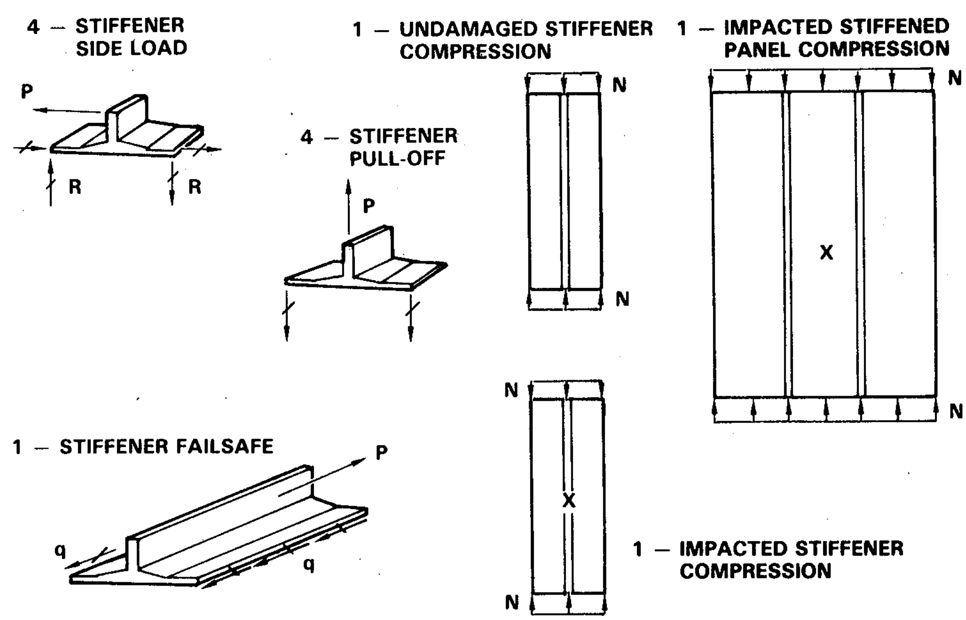


Figure 9

DESIGN DEVELOPMENT TESTS - STIFFENER
PULL-OFF AND SIDE LOAD TEST SET-UPS

The specimens used for the pull-off and side load tests were 3 inches long and 5.75 inches wide. The tests were conducted using a 50,000 lb MTS tensile test machine. In each case, the test load was applied along the top of the stiffener and reacted along the edges of the skin, as shown in Figure 10.

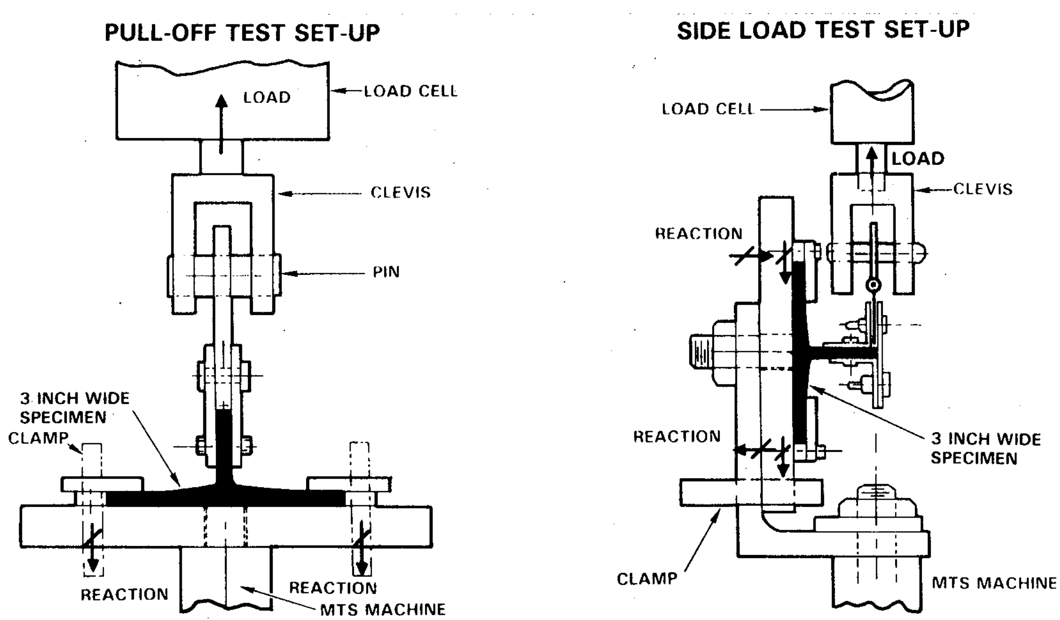


Figure 10

DESIGN DEVELOPMENT TESTS - STIFFENER PULL-OFF AND SIDE LOAD TEST RESULTS

The design requirement for each test was 300 pounds. Four specimens were tested in each load condition. The tests were conducted under room temperature, dry, conditions. The pull-off specimens failed at an average of 1687 pounds. The side load specimens failed at an average of 1257 pounds. In each case failure started as interlaminar cracking in the upstanding flange of the stiffener, below the line of fasteners through the stiffener. As the load was increased, the cracks propagated down to the base of the stiffener. Final failure in the pull-off specimens occurred as a delamination of the base of the stiffener just above the bond line to the outer skin. Final failure in the side load specimens occurred as multiple delaminations in the tapered flange of the stiffener on the side of the stiffener reacting the applied load in tension.

- **DESIGN REQUIREMENT**
  - **300 LB LOAD APPLIED OVER 3 INCHES**
- **TEST RESULTS**

TEST & CONDITION	SPECIMEN ID	FAILURE LOAD (LB)
PULL-OFF TEST 75°F, DRY	C1	1711
	C2	1862
	C3	1638
	C4	1536
	AVERAGE	1687
SIDE LOAD TEST 75°F, DRY	D1	1191
	D2	1309
	D3	1220
	D4	1309
	AVERAGE	1257

Figure 11

DESIGN DEVELOPMENT TESTS - STIFFENER
FAILSAFE TEST SET-UP

The failsafe specimen was 18 inches long and 5.75 inches wide. The test set-up was designed to load the joint between the stiffener and the surrounding panel in the same manner that it would be loaded if a stiffener in a wing were broken under load. The test was conducted under room temperature, dry, conditions. The test load was applied to the stiffener by loading plates extending the length of the stiffener. The load was reacted by a frame bolted to the areas of skin to either side of the stiffener. The specimen failed interlaminarly in the base of the stiffener as shown in the photo in Figure 12.

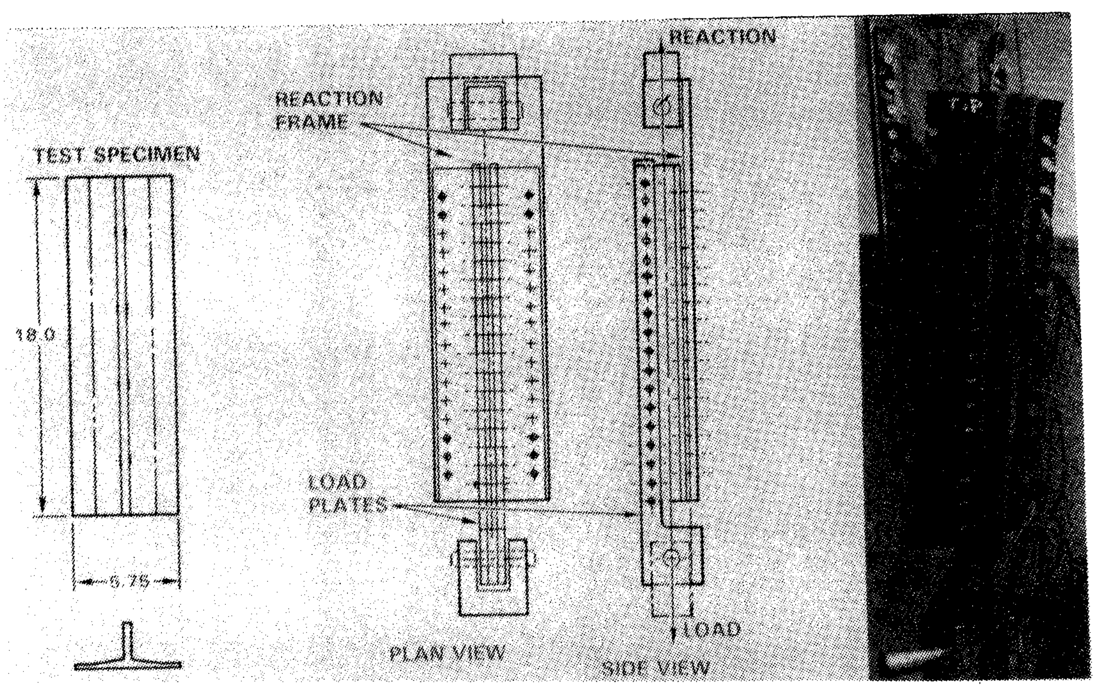


Figure 12

DESIGN DEVELOPMENT TESTS - STIFFENER
FAILSAFE TEST RESULTS

The design requirement for the failsafe specimen was 54,500 pounds. This was determined based on the specimen width of 5.75 inches, the design axial load intensity of -12,972 lb/in and 70 percent of the axial load being distributed in the stiffener. The specimen failed at 57,870 pounds.

- **DESIGN REQUIREMENT**
  - **SHEAR OUT THE LOAD FROM A BROKEN STIFFENER IN 18 INCHES, THE LENGTH OF 3 TIMES THE STIFFENER SPACING**

REQUIREMENT 54,500 LB

- **TEST DATA**

FAILURE LOAD 57,870 LB

Figure 13

DESIGN DEVELOPMENT TESTS - STIFFENER COMPRESSION TEST RESULTS

Undamaged and impact damaged stiffener specimens were tested in compression. Each specimen was 18 inches long and 5.75 inches wide. The specimens were reinforced on the ends with steel boxes and potting compound. The damaged specimen was impacted with 40 ft-lb, by a 12 pound impactor having a 0.5 inch hemispherical steel tup. The impact was made in the center of the specimen on the side of the stiffener 1.25 inches from the edge of the upstanding flange. Each specimen was instrumented with back-to-back strain gages on the skin surface, on the side of the stiffener, and on the top and bottom of the stiffener. The tests were conducted under room temperature, dry, conditions. The undamaged specimen failed at -189,500 pounds and a maximum strain level of -8783μ in/in. The impact damaged specimen failed at -178,500 pounds and a maximum strain level of -8505μ in/in.

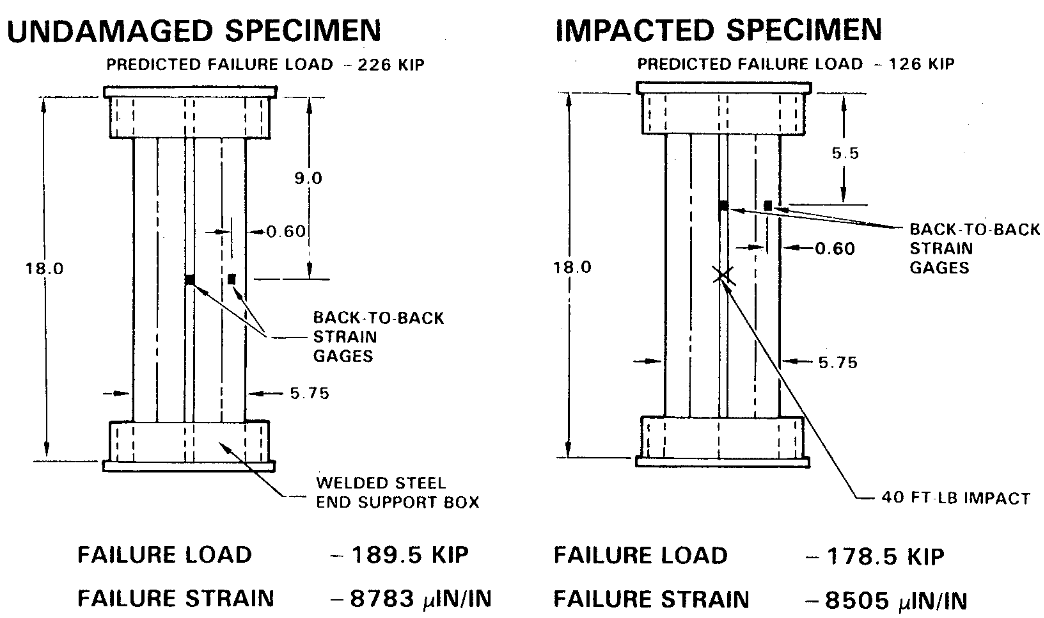


Figure 14

DESIGN DEVELOPMENT TESTS - STIFFENER COMPRESSION TESTS

Both specimens failed in combined compression and bending near the end of each specimen. The undamaged stiffener failed in the skin and in the body of the stiffener. The impacted specimen failed in the body of the stiffener near the top of the specimen as shown in Figure 15.

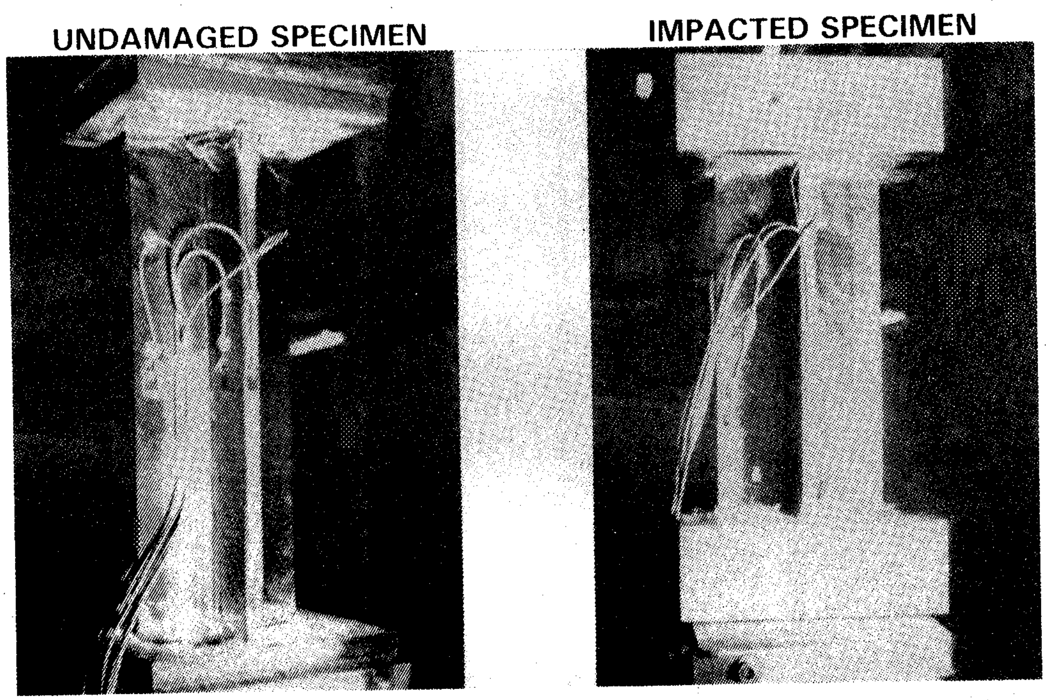
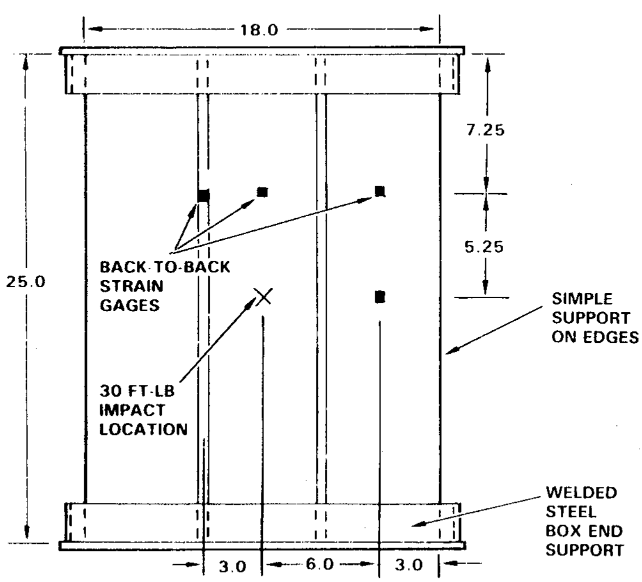


Figure 15

DESIGN DEVELOPMENT TESTS - STIFFENED
PANEL COMPRESSION TEST RESULTS

One 25 inch long by 18 inch wide two stiffener panel was tested in compression. The panel was first impacted in the center, between the stiffeners, with 30 ft-lb by a 12 pound impactor having a 0.5 inch diameter hemispherical steel tup. The impact caused 4.3 square inches of internal damage as measured by ultrasonic C-scan. The panel was instrumented with back-to-back strain gages and the ends were reinforced with steel boxes and potting compound. Steel angles were clamped to the free edges during the compression test to prevent buckling. The test was run under room temperature, dry, conditions. The panel failed at a load of 232,900 pounds, and a maximum strain of -5343 μ in/in.

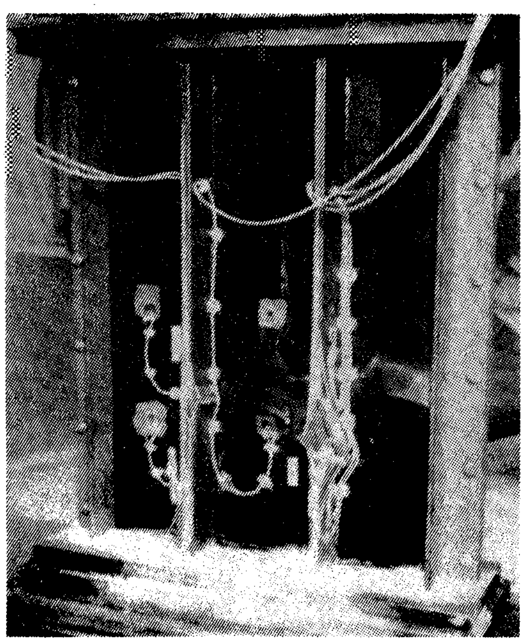


- **TEST RESULTS**
  - TEST CONDITION — 75°F, DRY
  - IMPACT ENERGY — 30 FT-LB
  - IMPACT C-SCAN DAMAGE AREA — 4.3 IN^2
  - PREDICTED FAILURE LOAD — -239 KIP
  - FAILURE LOAD — -232.9 KIP
  - FAILURE STRAIN — -5343 μIN/IN

Figure 16

DESIGN DEVELOPMENT TESTS - STIFFENED
PANEL COMPRESSION TEST RESULTS

The panel failed in combined axial compression and bending. The failures in the stiffeners are similar to those seen in the stiffener compression tests.



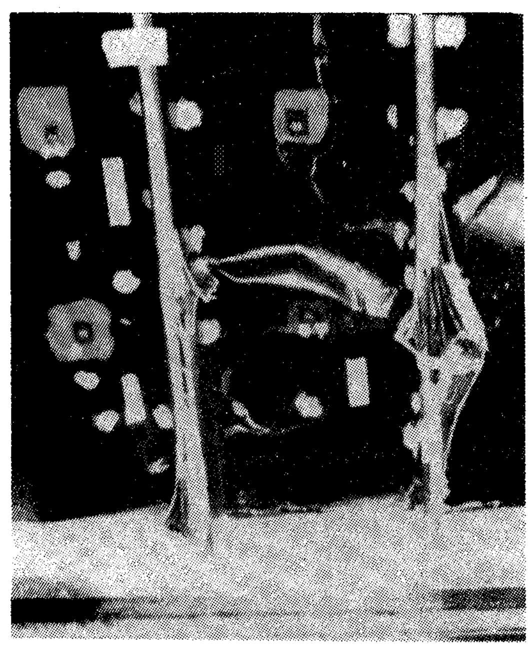


Figure 17

TECHNOLOGY DEMONSTRATION ARTICLE TEST OBJECTIVES

The objectives of the demonstration article tests were to validate the results of the previous lightning strike panel tests, fuel containment joint tests, impacted laminate fuel sealing coupon tests, and damage tolerance design studies by applying the technology developed to a structure representative of a transport wing.

- **LIGHTNING STRIKE PROTECTION**
  - NO BURN THROUGH
  - NO INTERIOR SPARKING
  - MINIMUM STRUCTURAL DAMAGE
- **FUEL CONTAINMENT**
  - JOINT SEALING
  - IMPACTED LAMINATE FUEL SEALING
- **DAMAGE TOLERANCE**
  - ULTIMATE LOAD CAPABILITY WITH NOTCHES AND BARELY VISIBLE IMPACT DAMAGE

Figure 18

TECHNOLOGY DEMONSTRATION ARTICLE TEST PLAN

The demonstration article will be tested to validate the technology developed during the first phase of the program. The panel will be struck with a simulated Zone 2 lightning strike of 100,000 amperes. To verify the techniques developed for fuel sealing, the panel will be impacted with 30 ft-lb to inflict barely visible damage to the outer surface, a fuel tank enclosure will be fitted, and a series of fuel pressure tests will be performed. The damage tolerance of the panel and attached substructure will be evaluated by applying one lifetime of axial fatigue ground-air-ground load cycles to the panel and then loading the demonstration article to failure in compression.

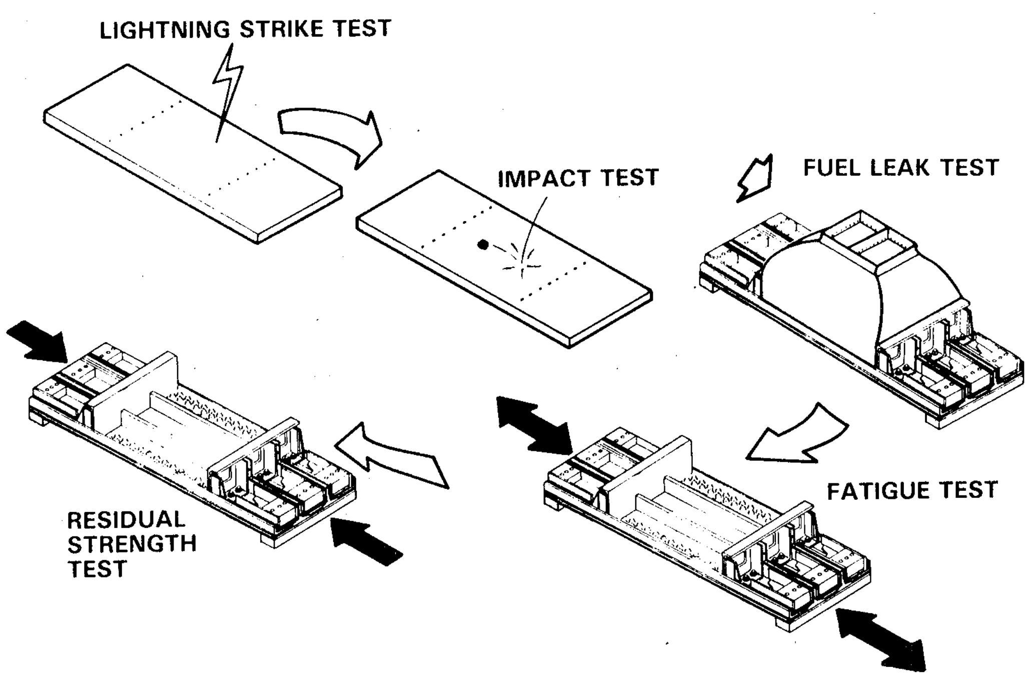


Figure 19

TECHNOLOGY DEMONSTRATION ARTICLE

The demonstration article was designed to represent a moderately loaded area of a 1990's transport aircraft wing. This structure was envisioned to be made up most likely of a graphite/epoxy wing spars and covers and aluminum substructure. The process development tooling was used to fabricate the integrally stiffened panel. Aluminum ribs were attached to the panel to represent the substructure-to-surface joint for the lightning strike test and to provide chord-wise support to the panel during the fuel pressure tests and the axial load tests. The each rib cap was machined from a standard aluminum extrusion and the three clips on each rib were machined from aluminum plate stock. Each part of the rib was painted prior to assembly. Graphite/epoxy spars were attached to the edges of the panel to represent the spar-to-cover joint for the fuel pressure tests and to support the edges of the panel during the axial load tests.

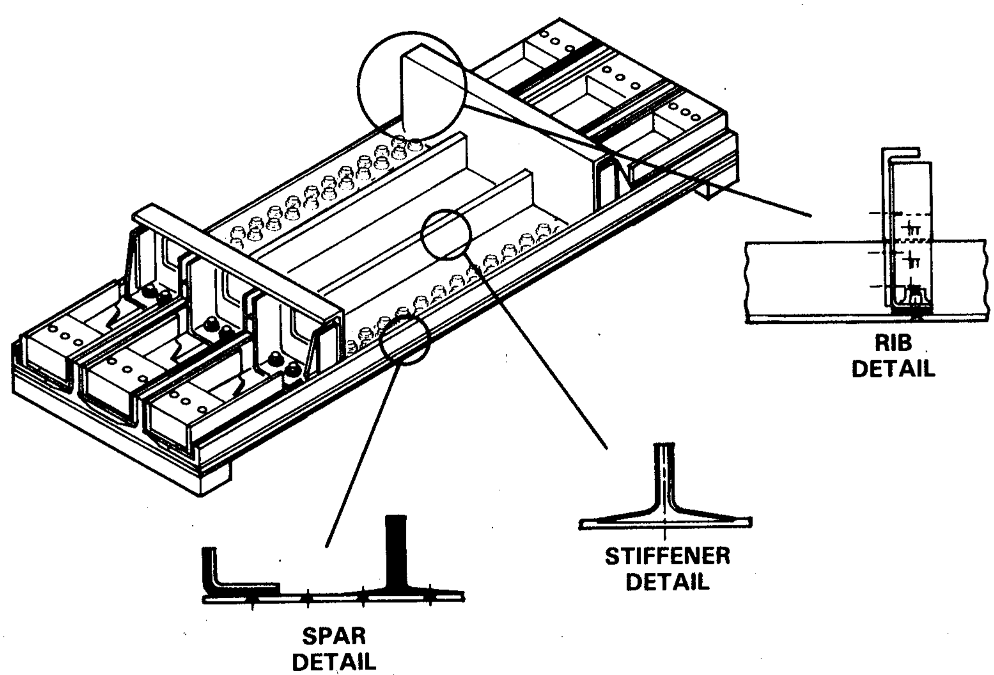


Figure 20

TECHNOLOGY DEMONSTRATION ARTICLE

A large aluminum enclosure was designed to mount to the backside of the demonstration article for the fuel pressure tests. The enclosure was attached to the aluminum ribs at each end, and to the graphite/epoxy spars on each side of the demonstration article in order to test the graphite/epoxy spar-to-stiffened panel skin joint and to test the joint between the stiffened panel and the aluminum rib, as shown below in Figure 21.

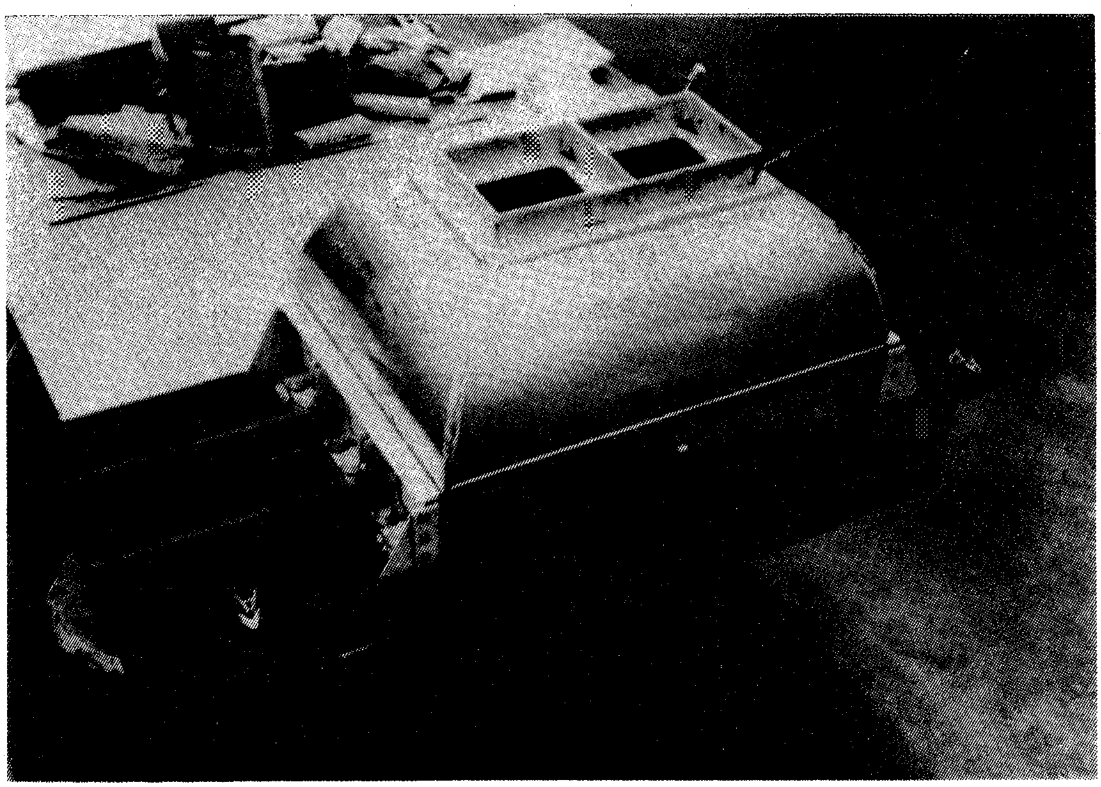


Figure 21

TECHNOLOGY DEMONSTRATION ARTICLE LIGHTNING STRIKE TEST SET-UP

The lightning strike test set-up used for the series of small panel tests was modified and used for the demonstration article test. In the test, the exterior surface of the demonstration article was struck, with a Zone 2 restrike of 100,000 amperes at 50,000 volts, along one line of fasteners attaching one of the aluminum ribs to the stiffened panel. A 130 knot stream of air was blown across the panel, simulating the airflow over a transport wing at approach speed.

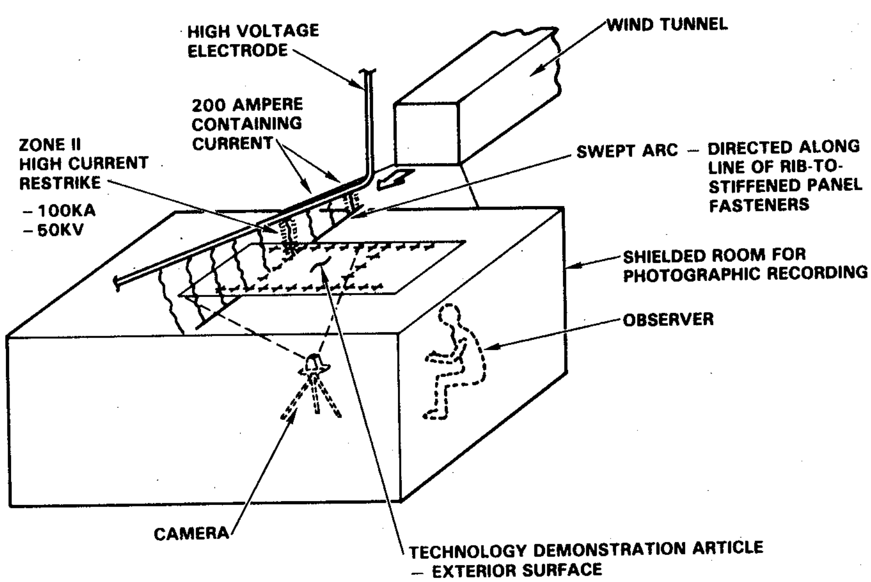
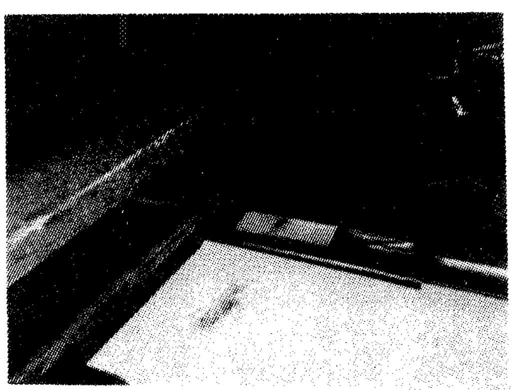


Figure 22

TECHNOLOGY DEMONSTRATION ARTICLE LIGHTNING STRIKE TEST RESULTS

The simulated Zone 2 lightning strike on the exterior of the demonstration article remove some exterior paint but resulted in no burn through or interior sparking. Ultrasonic C-scan of the panel indicated that no structural damage was done to the panel by the strike.



- **NO BURN THROUGH**
- **NO INTERIOR SPARKING**
- **MINIMUM STRUCTURAL DAMAGE**
  - **DAMAGE TO LIGHTNING STRIKE PROTECTION MATERIAL ONLY**

Figure 23

TECHNOLOGY DEMONSTRATION ARTICLE IMPACTED FUEL LEAK TEST

After the lightning strike test and ultrasonic inspection, the fuel tank enclosure was attached to the demonstration article and the assembly was proof tested, for leaks, at 10 psig. No leaks were encountered. The enclosure was then removed and the stiffened panel was impacted with 30 ft-lb, using a 0.5 inch diameter hemispherical steel tup attached to a 12 pound falling weight.

Tests to go include a 15 psig ultimate fuel pressure test, followed by one lifetime of ground-air-ground cycles and a second ultimate fuel pressure test.

- **PRESSURE TEST**
  - 10 PSI PROOF TEST
- **IMPACT**
  - 30 FT-LB IMPACT ON SKIN BETWEEN STIFFENERS
- **PRESSURE TEST**
  - 15 PSI ULTIMATE PRESSURE TEST
- **FATIGUE TEST**
  - 1 LIFETIME OF GROUND-AIR-GROUND CYCLES
- **PRESSURE TEST**
  - 15 PSI ULTIMATE PRESSURE TEST

Figure 24

TECHNOLOGY DEMONSTRATION ARTICLE AXIAL LOAD TESTS

Following the first ultimate fuel pressure test the demonstration article will be fatigue cycled for one lifetime of ground-air-ground cycles having a range ratio of R = -2.0, with barely visible impact damage on the skin. After the second ultimate fuel pressure test, the demonstration article will be loaded to failure in compression.

TEST	CONDITION	LOADS
FATIGUE	30 FT-LB SKIN IMPACT LIGHTNING STRIKE 75°F, DRY	36000 CYCLES 50% LIMIT LOAD 36 CYCLES 80% LIMIT LOAD
RESIDUAL STRENGTH	30 FT-LB SKIN IMPACT LIGHTNING STRIKE 75°F, DRY	LOAD TO FAILURE IN COMPRESSION

Figure 25

TECHNOLOGY DEMONSTRATION SUMMARY

Based on the tests conducted to date on the demonstration article, the lightning strike protection system consisting of 8 mil diameter aluminum wire/graphite fabric on the outer surface combined with sealant filled plastic caps applied over the fastener collars on the interior surface and painted aluminum substructure eliminates interior sparking and minimizes structural damage due to a Zone 2 lightning strike on a composite transport wing.

The fuel containment tests done to date have also confirmed the conclusion reached in the first phase of the program that the conventional fuel tank sealing techniques used for joints in metal structures are adequate for composite structures.

Future tests on the technology demonstration article will validate the effectiveness of the 0.005 in. thick coating of flexible polyurethane paint on the inside of the wing surface as an adequate method for preventing leaks due to low energy impact threats, demonstrate the damage growth inhibiting capability of the stiffened panel design and evaluate the ability of the demonstration article to withstand ultimate load with barely visible impact damage.

- **LIGHTNING STRIKE TEST**
  - PROTECTION SYSTEM CONSISTING OF 8 MIL DIAMETER ALUMINUM WIRE/GRAPHITE FABRIC AND SEALANT FILLED PLASTIC CAPS ELIMINATES INTERIOR SPARKING AND MINIMIZES STRUCTURAL DAMAGE
- **FUEL CONTAINMENT TEST**
  - CURRENT METAL WING STRUCTURE JOINT FUEL SEALING TECHNIQUES USING POLYSULFIDE SEALANT ARE ADEQUATE
  - CHEMGLAZE PAINT INTERIOR FUEL TANK SURFACE COATING IS REQUIRED TO PREVENT LEAKS DUE TO IMPACTS PRODUCING LESS THAN VISIBLE DAMAGE
- **DAMAGE TOLERANCE**
  - 5343 μIN/IN STRAIN (COMPRESSION) AT FAILURE OBTAINED WITH BARELY VISIBLE IMPACT TO THE SKIN

Figure 26

IMPACT DYNAMICS AND ACOUSTIC TRANSMISSION IN FUSELAGE STRUCTURES

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Burbank, California

ACEE Composite Structures Technology Conference
Third Special Oral Review
August 13-16, 1984

INTRODUCTION

This paper summarizes some of the pertinent results from the "Study of Utilization of Advanced Composites in Fuselage Structures of Large Transports" NAS1-17415 and discusses the follow-on program, "Transport Composite Fuselage Technology - Impact Dynamics and Acoustic Transmission" NAS1-17698.

The study program has been completed and the Final Reports will soon be released. The results of this program and similar ones at Boeing and Douglas lead to the initiation of three Transport Composite Fuselage Technology programs. This paper outlines the plans and approach to the Lockheed program.

The major focus of this program is to develop and demonstrate the impact dynamics and acoustic transmission technology for a composite fuselage which meets all design requirements of a 1990 large transport aircraft without substantial weight and cost penalties. The specific objectives are:

- To develop analytical methods for the behavior of advanced composite structures under crash conditions; to develop energy absorbent structural concepts and to verify the methodology and concepts by test.
- To develop analytical methods for the prediction of acoustic transmission through fuselage shells constructed from advanced composite materials and to verify by test.

A secondary objective of the program is to transfer the developed technology to the other U.S. transport aircraft manufacturers who are working on other key technology areas and to the U.S. composites industry.

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COMMERCIAL BASELINE ATX-350I

The baseline airplane is shown in Figure 1. It is an advanced technology version of the L-1011 aircraft. The structure is of conventional configuration and the primary materials are conventional aluminum alloys. The airplane has an advanced airfoil and an Æ of 12.00. Maneuver load control, dynamic gust response and fly-by-wire are included in the baseline. Advanced E^3 propulsion is included.

The fuselage is 203.5 feet long and 19.5 feet in diameter. The airplane is configured for 350 passengers and has a design range of 4600 nautical miles.

The barrel section considered during the study program and now being used for the current design studies is just behind the wing and is 20 feet long. This is the most highly loaded area of the fuselage and contains all the necessary structural features to evaluate fully the impact dynamics and acoustic transmission technologies and to incorporate other technology areas such as cutouts, joints, splices, pressure containment, damage tolerance and post-buckling.

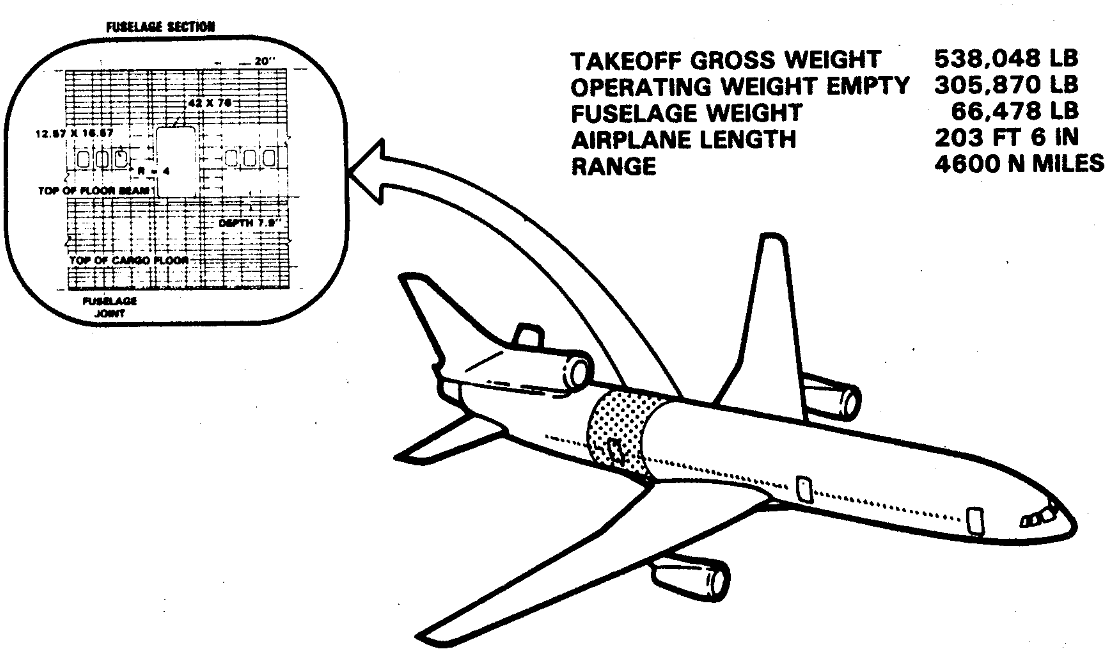


Figure 1

STRUCTURAL CONCEPTS

The study program identified several structural concepts which best meet all the diverse requirements associated with design and manufacturing. These concepts are illustrated in Figure 2. The skin stiffener concepts, blade and jay section stiffeners are structurally efficient and producible. Being open sections, they are also readily inspectable. The orthogrid concept which incorporates continuous blade stiffeners and blade frames is structurally less efficient but readily lends itself to automated fabrication and is a potentially low cost design.

The frame concepts are the orthogrid and a filament wound design which showed potential for further development. One outcome of the study was that development of frame fabrication techniques and innovative designs must be developed so that efficient frames can be designed and fabricated at a reasonable cost.

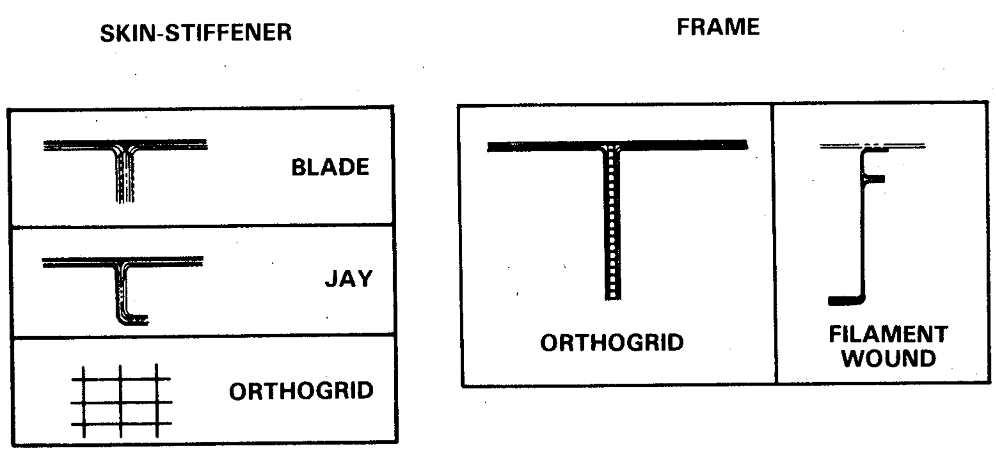


Figure 2

BENEFITS

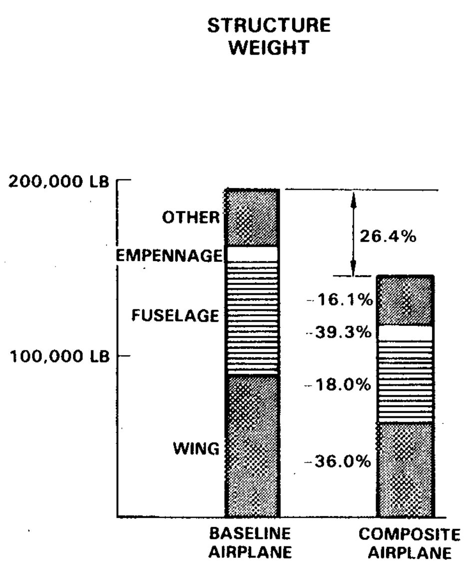
Part of the fuselage study program involved an assessment of the benefits of applying advanced composites to primary structures. Figure 3 shows the structure weight savings and the improved return on investment when advanced composites are incorporated. The composite airplane wing and empennage were resized to take advantage of the weight saved overall.

The fuselage volume must remain the same so only a small resizing is possible which is based on reduced wing and tail loads. The 26.4 percent weight saving was achieved with 73 percent use of composites in the fuselage, 77 percent in the wing and 59 percent in the empennage.

The return on investment analysis is based on an airline operating a fleet over a period of 16 years. A total of 8 airplanes are put in operation the first year, 8 more added in year 2 and 7 more in year 3 for a total fleet of 23 airplanes. The airplanes operate for 2500 mile average stage length with an annual utilization of 4162 hours. Fuel price is $1/gallon.

The two values shown for the all composite airplane represent the effects of automated fabrication. The lower value assumes minimum automation, the higher value major automation.

The operating cost saving is almost entirely due to reduced fuel consumption because of lower weights.



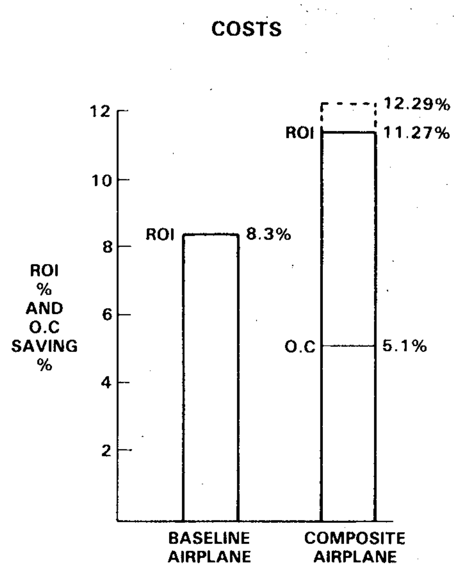


Figure 3

TECHNOLOGY ISSUES

A list of the technology issues was assembled from inputs of specialists in the various disciplines within Engineering, Manufacturing, and Quality Assurance. The issues identified were reviewed and ranked according to the urgency of their resolution. The issues are listed in Figure 4 in rough order of urgency.

The top six issues were rated of almost equal urgency because they all affect the basic design of a large transport fuselage. The lower six are important but their resolution is less urgent or the issue is being adressed elsewhere.

As a result of this assessment and similar ones by Boeing and Douglas, NASA Langley Research Center has funded three Transport Composite Fuselage Technology programs which address the top six issues specifically and to some extent all programs address some of the six issues.

Lockheed is developing the technology for impact dynamics and for acoustic transmission.

- IMPACT DYNAMICS
- ACOUSTIC TRANSMISSION
- JOINTS AND SPLICES
- PRESSURE CONTAINMENT
- POST BUCKLING
- SHELL CUTOUTS
- AUTOMATED MANUFACTURING
- PROCESSING SCIENCE
- ELECTROMAGNETIC EFFECTS
- REPAIR
- NDE/NDI
- FLAME/SMOKE

Figure 4

IMPACT DYNAMICS

Impact dynamics ranks as urgent because its resolution may affect the basic structural concepts of the lower fuselage structure. Figure 5 summarizes the state of the art with respect to advanced composite materials and structures under impact dynamics and lists the technology voids.

Metal structures have not generally been designed specifically for impact dynamics because the ductility and energy absorption capabilities of these structures have been deemed adequate. Because of the brittle nature of the high strength and stiffness advanced composites they do not absorb as much energy as metals.

The FAA currently requires that a composite structure be as good as its metal counterpart.

The state of the art involves specially designed crushable structures for the lower fuselages of helicopters. This approach for a large transport aircraft would prove costly. Predictive methodology is required so that the composite structural designs can be evaluated and the need for special treatments determined. If special treatments are required then inexpensive structurally sound concepts must be developed.

STATE OF THE ART

- MATERIAL CHARACTERIZATION AND CRUSHING TESTS HAVE DEMONSTRATED THAT ADVANCED COMPOSITE MATERIALS CANNOT ABSORB AS MUCH ENERGY AS METALS
- HELICOPTERS HAVE BEEN BUILT WITH SPECIAL ENERGY ABSORBENT LOWER FUSELAGE
- TEST DATA ALMOST NONEXISTENT FOR LARGE COMPOSITE STRUCTURES

TECHNOLOGY VOIDS

- DEVELOPMENT OF STRUCTURAL CONCEPTS FOR LARGE AIRCRAFT CAPABLE OF ABSORBING ENERGY AND EFFICIENTLY CARRYING STRUCTURAL LOADS
- PREDICITIVE METHODOLOGY FOR COMPOSITE STRUCTURES

Figure 5

ACOUSTIC TRANSMISSION

Acoustic transmission is considered urgent because the magnitude of the problem must still be defined. The significance of this issue is the relationship between the interior noise level and the mass of the fuselage shell. The state of the art and the technology voids are summarized in Figure 6.

Analytical methodology and interior treatments are available for metallic shells which keep interior noise ≤ 80 dBA. There is limited data for composite shells. Interior noise generated by turbofan engines is only a minor problem today and propeller noise is a localized problem which can be solved by localized treatments. The major source of interior noise for a composite shell is turbulent boundary layer generated.

The first technology void which must be filled is the development of analytical methods for the prediction of interior noise inside the anisotropic composite fuselage shell and verification of this methodology by test. Once the magnitude of the problem has been identified, then the remaining technology voids will be related to structural design concepts which help to reduce interior noise and effective interior treatments which will not add back much or all of the weight savings of composites over metals.

STATE OF THE ART	TECHNOLOGY VOIDS
• METHODS AVAILABLE FOR THE PREDICTION OF INTERIOR NOISE IN METALLIC FUSELAGES • INTERIOR TREATMENTS AVAILABLE TO REDUCE INTERIOR NOISE, FOR A WEIGHT PENALTY — INTERIOR NOISE LEVELS ⩽ 80 dBa • LIMITED TEST DATA AVAILABLE FOR COMPOSITE PANELS — NO TEST DATA ON COMPOSITE CYLINDERS	• ANALYTICAL METHODS FOR PREDICTION OF COMPOSITE STRUCTURE NOISE REDUCTION — PARTICULARLY BOUNDARY LAYER NOISE • EFFECTIVE TREATMENTS TO REDUCE INTERIOR NOISE IN A COMPOSITE FUSELAGE WHICH WOULD NOT NEGATE MOST OR ALL OF WEIGHT SAVINGS OF COMPOSITES OVER METALS

Figure 6

OBJECTIVES OF TRANSPORT COMPOSITE FUSELAGE TECHNOLOGY CONTRACT NAS1-17698

The objectives of this contract are summarized in Figure 7. The full title of the contract is "Transport Composite Fuselage Technology - Impact Dynamics and Acoustic Transmission" and it is part of the NASA Advanced Composite Structures Technology (ACST) program. The contract is intended to address long-lead-time critical composite technology for fuselage structure, primarily in the areas of impact dynamics and acoustic transmission.

The specific objective of this contract is to develop and demonstrate the technology for impact dynamics and acoustic transmission as related to a composite fuselage which meets all design requirements for a 1990's large transport aircraft without substantial weight and cost penalties.

A secondary objective of this contract is to transfer the technology developed to other U.S. transport aircraft manufacturers working on other critical technology areas of the ACST program. The other technology areas are being worked on by Boeing and Douglas: Boeing is under contract to develop the technology for pressure containment and damage tolerance, and Douglas is under contract to develop the technology for joints and cutouts.

OBJECTIVE:

DEVELOP AND DEMONSTRATE THE IMPACT DYNAMICS AND ACOUSTIC TRANSMISSION TECHNOLOGY FOR A COMPOSITE FUSELAGE FOR 1990'S LARGE TRANSPORT AIRCRAFT

Figure 7

SCOPE

The contract is divided into four technical phases. These phases are summarized in Figure 8.

Phase I is Design Optimization. Specific areas of the ATX 3501 fuselage were selected for design development relative to impact dynamics and acoustic transmission. A preliminary design study was performed. The design is being developed to incorporate energy absorbing and noise attenuation techniques. A baseline for a 5.5 foot diameter metal and composite fuselage section is being designed. The designs will be evaluated for technology readiness and producibility and the analytical methodology being developed will be assessed and modified based on test results.

Phase II, Methodology Development. The predictive analytical methodology for impact dynamics and acoustic transmission is being developed.

Phase III, Process Development Fabrication. A preliminary process development plan has been prepared using state-of-the-art materials and processes. Tooling will be designed and fabricated and test components will be fabricated.

Phase IV, Technology Demonstration. During this phase, development tests will be performed to evaluate the candidate designs so that the selection of a final design can be made.

- **PHASE I**
  - **DESIGN STUDIES OF BARREL SECTION**
  - **DESIGN TEST COMPONENTS**
  - **EVALUATE DEVELOPED TECHNOLOGY**

PHASE II

- **DEVELOP ANALYTICAL METHODOLOGY**

PHASE III

- **PROCESS DEVELOPMENT**
- **FABRICATE TEST COMPONENTS**

PHASE IV

- **DEVELOPMENT TESTS**
- **DEMONSTRATION TESTS**

Figure 8

PROGRAM MASTER SCHEDULE

The program master schedule is depicted in Figure 9.

The contract go-ahead was April 26, 1984. The period of performance is 32 months. The technical effort is scheduled to be completed at the end of June, 1986. Following the completion of the technical effort, a Technology Demonstration Review will be held followed one month later by a Final Oral Review at NASA Langley Research Center.

Throughout the program workshops will be held approximately every six months. These workshops will be held with the other fuselage technology contractors and representatives from NASA. The primary objective of these workshops is to facilitate technology transfer.

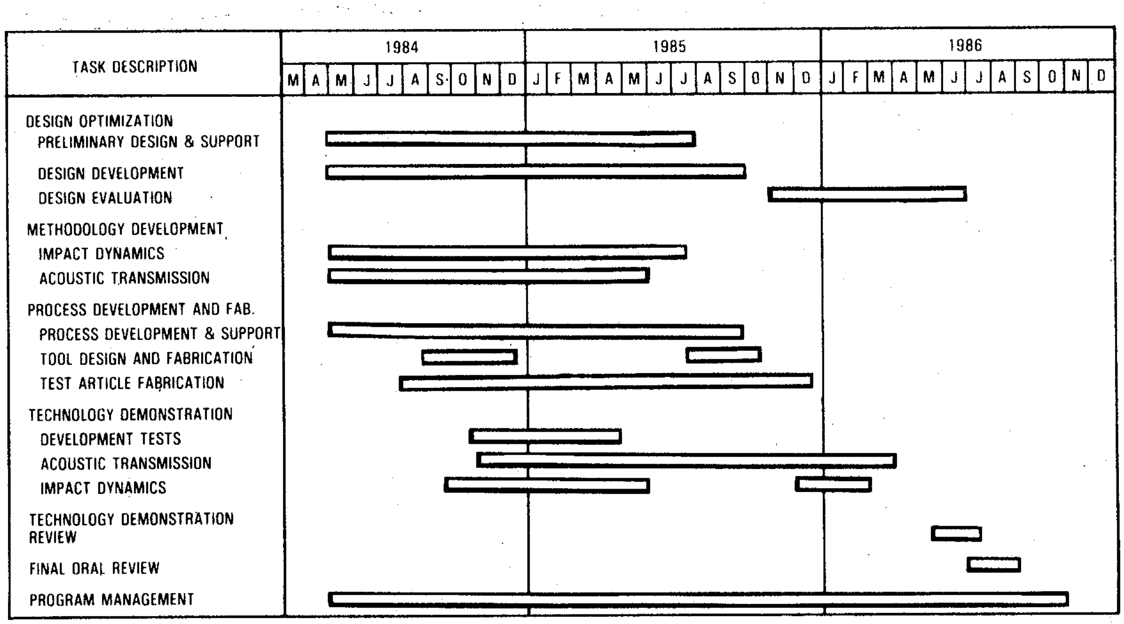


Figure 9

IMPACT DYNAMICS OBJECTIVES

The primary objectives of this portion of the program are to develop analytical methods for the behavior of advanced composite structures under crash conditions, to develop energy absorbent structural concepts, and to verify the methodology and concepts by test. (See Figure 10)

Analytical and test comparisons will be made.

OBJECTIVES

- **DEMONSTRATE CAPABILITY**

  DESIGN

  METHODOLOGY

- **COMPARISON BETWEEN CURRENT METAL AND ADVANCED MATERIAL DESIGNS**

- **OBTAIN IMPACT LOAD DATA**

Figure 10

FAA GUIDELINES

In order to satisfy the FAA guidelines shown in Figure 11, advanced material designs must be as good as current metal designs. The current crash design requirements for transport airplanes which affect fuselage design are stated in Federal Aviation regulations: "Part 25 - Airworthiness Standards: Transport Category Airplanes" paragraph 25.561, which is summarized below:

The airplane, although it may be damaged in emergency landing conditions on land or water, must be designed as prescribed to protect each occupant under those conditions. The structure must be designed to give each occupant every reasonable chance of escaping serious injury in a minor crash landing when:

(1) Proper use is made of seats, belts, and all other safety design provisions.

(2) The wheels are retracted (where applicable), and

(3) The occupant experiences the following ultimate inertia forces acting separately relative to the surrounding structure: Upward - 2.0g, forward - 9.0g, sideward - 1.5g, downward - 4.5g, or any lesser force that will not be exceeded when the airplane absorbs the landing loads resulting from impact with an ultimate descent velocity of five fps at design landing weight.

The supporting structure must be designed to restrain, under all loads up to those specified in paragraph (b) (3) of this section, each item of mass that could injure an occupant if it came loose in a minor crash landing.

A. IMPACT DYNAMICS

THE PRESENT APPROACH IN AIRFRAME DESIGN IS TO ASSURE THAT OCCUPANTS HAVE EVERY REASONABLE CHANCE OF ESCAPING SERIOUS INJURY UNDER REALISTIC AND SURVIVABLE IMPACT CONDITIONS. EVALUATION MAY BE BY TEST OR BY ANALYSIS SUPPORTED BY TEST EVIDENCE. TEST EVIDENCE INCLUDES BUT IS NOT LIMITED TO ELEMENT OR SUBCOMPONENT TESTS AND SERVICE EXPERIENCE. ANALYTICAL COMPARISON TO CONVENTIONAL STRUCTURE MAY BE USED WHERE SHOWN TO BE APPLICABLE.

Figure 11

IMPACT DYNAMICS APPROACH

The impact dynamics approach is illustrated in Figure 12. An assessment will be made of the baseline fuselage to determine the overall behavior and to identify the regions of the fuselage which require investigating for the incorporation of energy absorbing structure. This assessment will be accomplished using a computer program called "KRASH" which is described in FAA-RD-77-189 Volumes I, II, III "KRASH User's Manual," Gamen, M. A., Wittlin, G.

Methodology development includes three subtasks; technology assessment, analysis of energy absorbing designs, and derivation of special design features. The technology assessment incorporates available test and analysis data, as well as on-going and planned activities that will contribute pertinent information to this program. The design of energy absorbing concepts requires the formulation of analytical procedures which will be experimentally verified. The derivation of special design features is based on the expectation that individual elements will contribute load and energy absorption capability in different quantities.

The development tests will consist of fabrication and testing of different structural elements that comprise the fuselage lower shell segment. Prior to testing, the various design concepts will be analyzed. Test and analysis results will be compared. The analysis procedures will be refined to incorporate empirical factors, as required. Comparisons will also be made between metal and advanced material designs for the same design function and load condition.

The demonstration test will consist of one test article which incorporates the desirable design features of the previously tested development structural elements. Analysis to predict dynamic responses will be performed prior to testing.

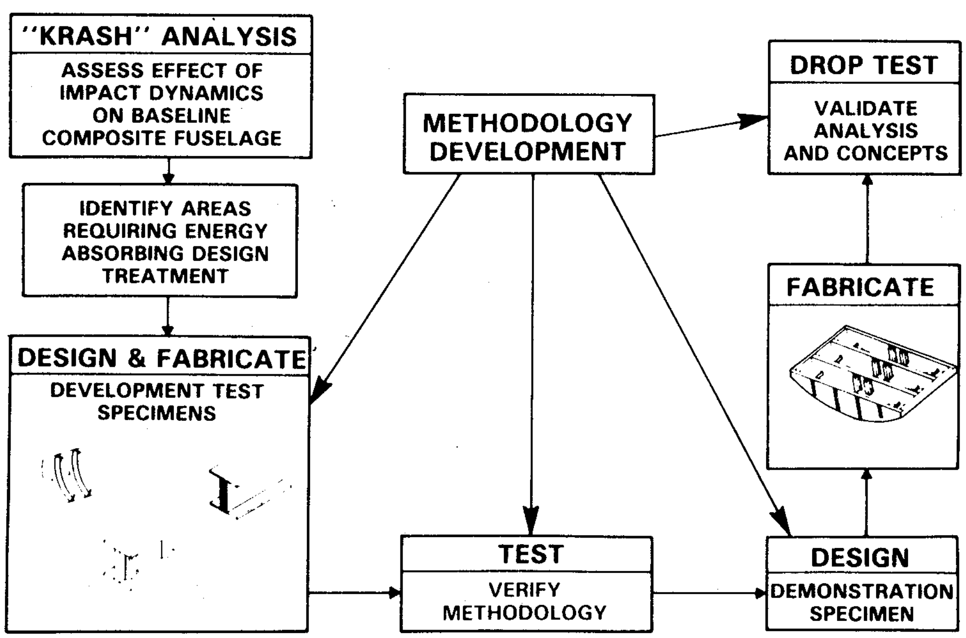


Figure 12

IMPACT TEST SETUP

An impact test machine will be designed which will allow the energy absorption demonstration article (a lower fuselage segment containing a 60 by 100 inch cargo floor) to be guided in a horizontal position attitude during a free-fall drop to impact a reaction load platform at a prescribed velocity of 10 ft/sec. The impact test machine, as shown in Figure 13 will consist of a carriage assembly, a reaction platform, and a guide-support frame.

The carriage assembly is a frame with a set of rollers at each of eight corners that ride on the guide support frame to allow the carriage to travel only in the vertical direction. The carriage assembly device function is to transport the test article from a predetermined drop height free-fall to a reaction platform. The reaction platform assembly consists of a stiffened plate structure instrumented with five load cells (force transducers), four load cells located at the bottom of the platform at each of the four corners and one load cell at the center. The reaction platform will be installed on a massive concrete floor (at least 4 feet thick) that exists in the test laboratory. The load cells will be installed to measure reaction loads to the high impedance floor.

The guide-support frame will be constructed of existing standard laboratory frame set. The guide-support frame will have a remote control release hook that engages a chamfered head bolt on the carriage when the carriage and specimen are raised to predetermined drop height.

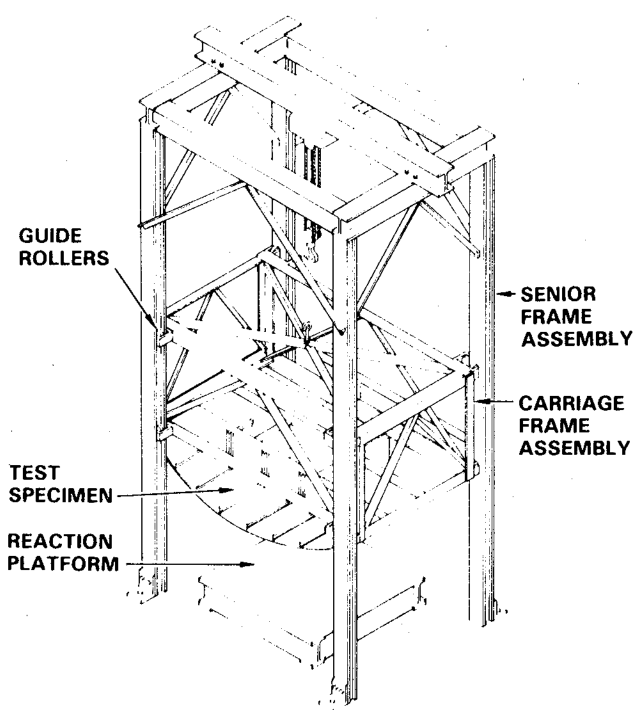


Figure 13

ACOUSTIC TRANSMISSION OBJECTIVES

A primary objective of the acoustic transmission portion of the program is to develop a reliable interior noise prediction analysis method for advanced composite fuselage shells and to validate this methodology by testing a composite cylinder. (see Figure 16.)

The other primary objective is to determine the fuselage structural configuration which with interior trim panels will meet the interior noise level requirements with the minimum weight penalty.

- **DEVELOP A RELIABLE INTERIOR NOISE PREDICTION METHOD**
- **MINIMIZE FUSELAGE STRUCTURAL WEIGHT PENALTIES**

Figure 14

ACOUSTIC TRANSMISSION APPROACH

The acoustic transmission approach is illustrated in Figure 15. A baseline aluminum and a baseline composite cylinder are being designed to strength and stiffness criteria. These cylinders are 5.5 feet in diameter and are similar to the Swearington Metroliner fuselage which was used as the test bed in an earlier noise study.

The methodology development starts with an evaluation of current analytical techniques for the prediction of aircraft interior noise. This is followed by the theory most appropriate for composite cylinder analysis and the coding of the necessary computer programs.

The methodology is then used to predict the interior noise levels in the two baseline cylinders and to determine the interior treatments required to reduce the interior noise to acceptable levels. Structural changes will be recommended where possible.

An optimum composite cylinder will then be designed, analyzed and fabricated for validation testing to demonstrate the technology developments. The methodology will be refined as required based on the test results.

Finally, a parametric design study of the ATX 350I fuselage and interior trim will be made to identify the most promising structural and trim configurations to provide an interior noise environment comparable to current wide body turbofan aircraft.

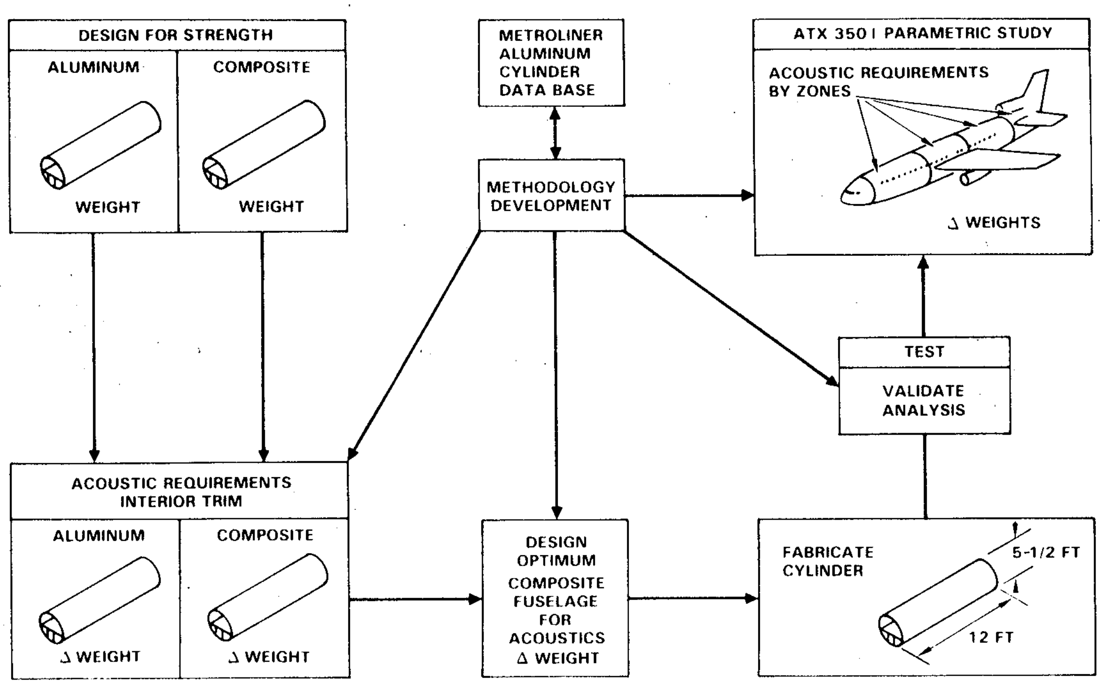


Figure 15

ACOUSTIC TEST SETUP

The composite cylinder will be evaluated for acoustic and simulated turbulent boundary layer excitations which will require two different test arrangements. Figure 16 shows on the left the setup for the acoustic excitation tests. The close-ended specimen will be mounted in a horizontal position in an anechoic chamber about two feet above the floor with the supports located at the extremities of the cylinder. The test article will be untreated. A noise generator coupled to an exponential horn will provide broadband random excitation from, 100 to 1,500 Hz. It will be positioned at normal incidence and at a distance from the surface of the cylinder which will provide the required sound distribution (the exact location of the noise source relative to the cylinder will be established from free field measurements in the anechoic chamber without the presence of the test article).

Figure 16 shows on the right the test arrangement for the simulated turbulent boundary layer excitation tests. The cylinder will be enclosed with a lead vinyl shroud and the excitation will be provided with multiple speakers coupled to the cavity between the shroud and cylinder walls. At the opposite end, acoustical foam will be installed to eliminate sound reflections over the frequency range of interest. Provisions will be made for the introduction of carbon dioxide into the cavity to create the desired mismatch in the exterior and interior speed of sound which exists under actual flight conditions. In this case, as shown, the test article will be in a vertical position which facilitates the introduction of the CO_2 gas, as well as the installation of the lead vinyl shroud. The speed of sound in the cavity will be determined using correlation techniques. This will require two microphones, one upstream and one downstream of the noise source.

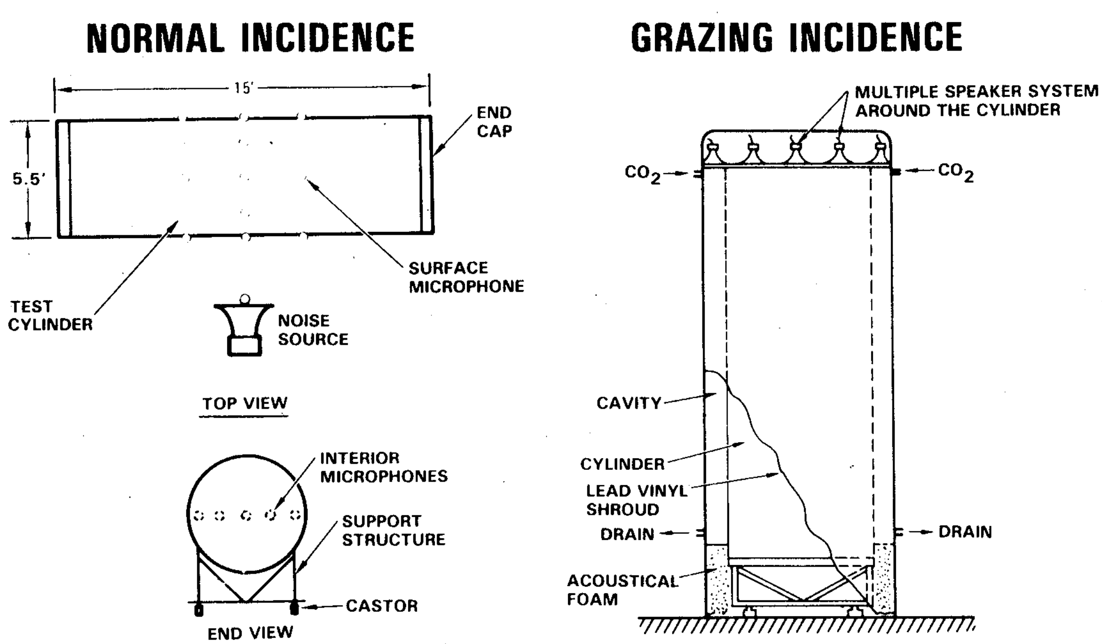


Figure 16

SUMMARY OF NOISE ATTENUATION TESTS

The noise attenuation test program is summarized in Figure 17. Modal studies of the bare and trimmed cylinder will define modal frequencies, modal damping, and mode shapes for the cylinder and the small skin panels bounded by stiffeners.

The sound transmission tests will be conducted for two different excitation fields, over the frequency range of 100 to 1500 Hz; i.e., broadband random acoustic excitation and simulated turbulent boundary layer excitation. The following tests will be performed on the composite cylinder.

- Normal incidence acoustic tests in an anechoic chamber without the presence of the shroud.
- Grazing incidence acoustic tests with air in the cavity between the shroud and cylinder walls.
- Grazing incidence acoustic tests with carbon dioxide in the shroud cavity.

In addition to the above tests, free-field sound measurements will be made in the anechoic chamber without the presence of the test article to determine the directivity and frequency characteristics of the noise source. A single noise generator coupled to an exponential horn will provide the excitation for the normal incidence tests. Measurements will be made at several excitation levels. In contrast, for the grazing incidence tests 8 to 12 speakers will be used to drive the shroud cavity. For this multiple speakers system, the acoustic output of the individual sources will be maintained constant and uncorrelated to simulate more closely the boundary-layer field in the circumferential direction.

TESTS	TEST ARTICLE	TEST CHAMBER	EXCITATION	DATA
1. MODAL ANALYSIS	BARE COMPOSITE CYLINDER	ANECHOIC	IMPACT	STRUCTURAL RESONANCES, MODES SHAPES AND DAMPING
2. FREE FIELD	NONE (EMPTY CHAMBER)	ANECHOIC	SINGLE NOISE SOURCE	SOURCE CHARACTERISTICS (FREQUENCY RESPONSE, LEVEL AND DIRECTIVITY)
3. NORMAL INCIDENCE SOUND TRANS-MISSION	BARE COMPOSITE CYLINDER	ANECHOIC	SINGLE NOISE SOURCE	● SURFACE PRESSURE FIELD ● INTERIOR SOUND FIELD ● STRUCTURAL RESPONSE ● NOISE REDUCTION
4. GRAZING INCIDENCE SOUND TRANS-MISSION	SHROUD ENCLOSED CYLINDER WITH AIR IN CAVITY	N.A.	MULTIPLE SPEAKER SYSTEM	SAME AS 3 ABOVE
5. SIMULATED BOUNDARY LAYER SOUND TRANS-MISSION	SHROUD ENCLOSED CYLINDER WITH CO_2 IN CAVITY	N.A.	MULTIPLE SPEAKER SYSTEM	SAME AS 3 ABOVE

Figure 17

PROGRESS

The progress to date is summarized in Figure 18.

Impact Dynamics

An analysis of the metal baseline fuselage using computer program "KRASH" showed that the major crushing of lower fuselage structure in a 10 ft/sec impact with the ground at 163 KEAS forward speed occurs in the aft fuselage. In a nose down or level attitude the nacelles hit the ground first and forward fuselage crushing is relatively small because the airplane rotates rapidly into a nose up attitude causing the aft fuselage to impact.

A KRASH analysis of a typical fuselage frame indicates that most deformation and energy absorption occurs in the lower segment between the floor post (approximately 90 percent). There is a significant difference between wide body and narrow body in this respect.

Design of development test specimens representing frames and bulkheads is progressing. Methodology development is underway.

Acoustic Transmission

The methodology development is well underway with some elements of the computer program up and running.

Design of the baseline metal and composite 5.5 foot diameter cylinders is well advanced.

IMPACT DYNAMICS

- **LOWER AFT FUSELAGE IDENTIFIED AS MOST CRITICAL**
- **METHODOLOGY BEING DEVELOPED**
- **DEVELOPMENT TEST SPECIMENS IN DESIGN**

ACOUSTICS

- **METHODOLOGY BEING DEVELOPED**
- **BASELINE ALUMINUM AND COMPOSITE CYLINDERS IN DESIGN**

Figure 18

SUMMARY

- A STUDY PROGRAM HAS BEEN COMPLETED DEFINING THE TECHNOLOGY ISSUES FOR LARGE TRANSPORT COMPOSITE FUSELAGES

- WORK IS NOW IN PROGRESS TO DEVELOP TECHNOLOGY IN THE FIELDS OF IMPACT DYNAMICS AND ACOUSTIC TRANSMISSION

Figure 19

COMPOSITE TRANSPORT WING TECHNOLOGY DEVELOPMENT
PROGRAM OVERVIEW-COVER/RIB CONCEPTS

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ACEE Composite Structures Technology Conference
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INTRODUCTION

The driving force behind the application of advanced composite materials to aircraft structures is the potential for significant weight savings and possible cost reduction. The application of advanced composites in redesigned secondary and medium primary components of large aircraft have demonstrated structural weight savings of 22 to 29 percent and cost savings of 10 to 12 percent over their metal counterparts, Therefore, the potential weight savings obtained by use of composite materials are substantial, especially when an aircraft is initially designed for application of this material. This weight savings can provide improved performance and/or reduced fuel requirements. By extending the application of advanced composites to wing and fuselage components, which comprise about 70 percent of the aircraft structural weight, 25 percent aircraft structural weight reduction and a corresponding 15 percent fuel savings should be achievable.

As a follow-on to the ACEE programs, NASA established the Advanced Composite Structures Technology (ACST) program to develop a composite primary airframe structures technology base to achieve the full potential of weight, fuel, and cost savings possible for future civil and military transport aircraft. As part of the ACST program, this Composite Transport Wing Development Program was initiated in 1984. In this program a composite center wing box that meets the design requirements for an advanced C-130 transport will be developed and evaluated in ground tests under NASA contract NAS1-17699.

The center wing box of the C-130 transport is of sufficient size and complexity to fully interrogate the many engineering and manufacturing technology issues which must be resolved before composite wing structures can be confidently applied to large transport aircraft.

The duration of this program is 64 months, with completion scheduled for August 1989. The monetary value of the contract is 26.1 million dollars.

This program is being conducted jointly by the Lockheed-California Company and the Lockheed-Georgia Company. The discussion of program activities is presented in two parts, with A.M. James providing a full program overview with a detail description of the program activities that the Lockheed-California Company is responsible for and then Mr. W.E. Harvil follows this presentation with a discussion of the program activities for which the Lockheed-Georgia Company is responsible.

LONG-TERM OBJECTIVE

Engineering and manufacturing technolgies as related to advanced composite materials have not been developed to the level required for their application to large primary aircraft structures. A comprehensive study was completed for the National Aeronautics and Space Administration (NASA), Langley Research Center in 1978 which outlined technology development plans for the utilization of advanced composites in commercial aircraft wing primary structure. That plan is being implemented through this program. The long-term objective of this program is to develop and demonstrate that engineering and manufacturing technology is available for application of advanced composite materials to primary wing structure of future aircraft.

- **DEVELOP ENGINEERING AND MANUFACTURING TECHNOLOGY READINESS FOR THE APPLICATION OF ADVANCED COMPOSITE MATERIALS TO LARGE WING STRUCTURE ON AIRCRAFT INTRODUCED IN THE LATE 1980 – EARLY 1990 TIME PERIOD**

Figure 1

SELECTION CONSIDERATIONS

The aircraft selection considerations included those listed here, in addition to the potential for cost and weight savings. It is desirable to have a production or derivative aircraft which could be used in meeting late 1980's requirements, from the standpoint of possible incorporation into production, in a timely fashion. The selected A/C must be representative of large transport technology and load/mission requirements, with a component at production interfaces which is affordable within the development program cost constraints. It should be generic in the sense that it allows for addressing key technology issues and provides insight into probable performance improvements in areas such as payload/range, field length, and fuel consumption.

- **PRODUCTION/DERIVATIVE AIRCRAFT**
- **MEET LATE 1980'S REQUIREMENTS**
- **TIMELY – REPRESENTATIVE**
- **LARGE BUT AFFORDABLE**
- **GENERIC**

Figure 2

LOCKHEED C-130 AIRPLANE

The C-130 airplane is a multiengine, long-range, cargo and troop carrier transport, constructed of aluminum, steel and titanium materials, powered by four propjet engines, and operated by a four-man crew.

Since Lockheed started operating Air Force Plant No. 6 in 1951, over 1700 C-130 aircraft have been produced for the Air Force, Navy, Coast Guard, foreign nations, and domestic commercial operators. Improvements have been made over the years, and will continue to be made, to increase the C-130's versatility to perform a multitude of missions. These include uprated power plants, the latest in electronics systems, improved wing structures and use of new materials. Manufacturing technologies have also been introduced into the manufacturing plan and include automatic riveting of wing beams and panels.

The Company is presently producing C-130 aircraft at a rate of 3 per month, with projections to maintain this production rate through the 1990's.

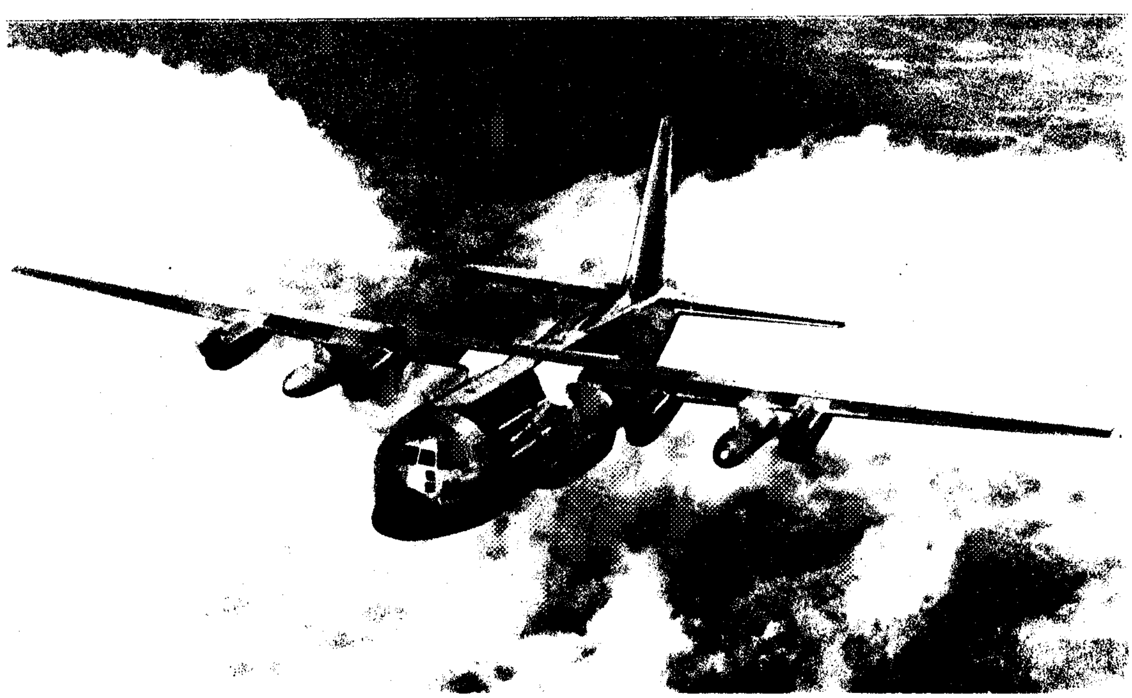


Figure 3

CANDIDATE AIRCRAFT

A number of aircraft in Advanced Design were initially surveyed to select four prime candidates for composite wing development. These candidates included the current C-130H aircraft and its commercial production counterpart, the L-100-30. For the missions expected in the late 1980's and 1990's, advanced derivatives of these were considered. Figure 4 compares the basic C-130H with an Advanced C-130, and with its commercial counterpart, the L-100-50Z. The two aircraft are structurally similar except for the fuselage length, where the commercial version is 35 feet longer, and where center wing fuel is not required.

	C-130H	ADVANCED C-130	L-100-50Z (COMMERCIAL)
GROSS WEIGHT – LB.	155,000	180,000	175,000
OPER. WT. EMPTY – LB.	79,210	90,500	85,180
ZERO FUEL WT. – LB.	99,140	134,500	152,010
LOAD FACTOR – G	3.0	3.0	2.5
PAYLOAD – LB	19,930	44,000	66,840
CENTER WING FUEL TANK	BLADDER	WET	NONE

Figure 4

C-130 CENTER WING-EXISTING METALLIC DESIGN

The component selected for the technology development is the center wing structural box of an advanced version of the Lockheed C-130 aircraft. The center wing has a constant cross section, zero dihederal and zero sweep. The existing structural box, shown in Figure 5, is a two-spar, multirib design, 440-inches long, 80-inches wide, and 35-inches deep at the crown. As noted on the figure, the structure is built-up from over 2200 parts utilizing over 50,000 fasteners. The existing component is currently in production at the Lockheed-Georgia plant in Marietta, Georgia, and is operational on both military and commercial versions of the aircraft. This compliance with dual military and commercial requirements will be maintained throughout the program.

The configuration of the composite wing box assembly will be similar to that of the metal wing but the composite box will be designed to the higher loading requirements of the Advanced C-130 aircraft.

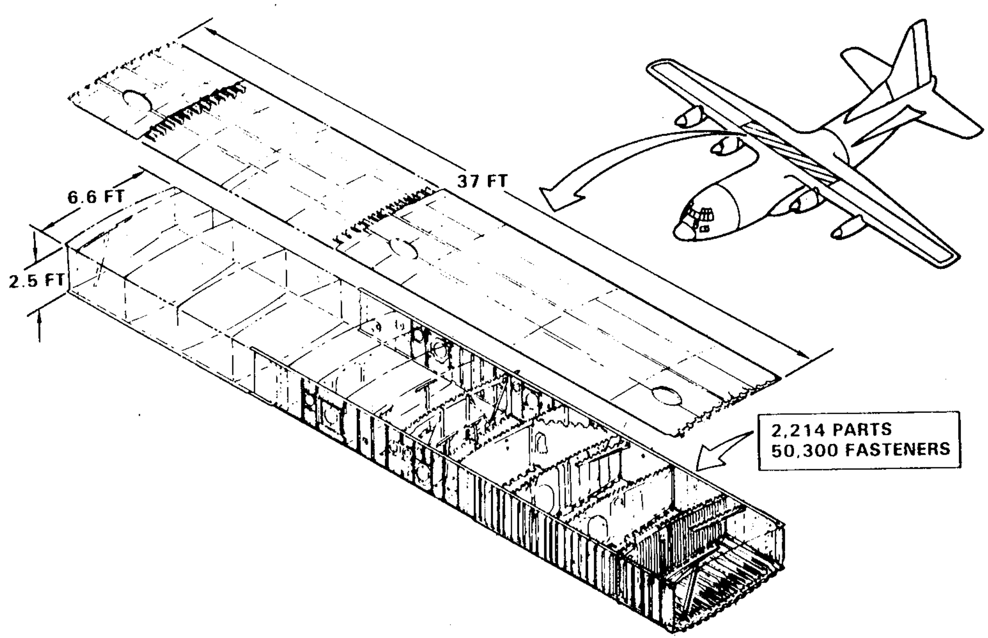


Figure 5

C-130 CENTER WING BOX ASSEMBLY

A photograph of the center wing box in the assembly fixture is shown in Figure 6. The center wing upper surface configuration is integrally stiffened skin and machined tapered hats. The lower surface is machine tapered base plate and extruded hats. The center wing has five access doors. Three are located on the upper surface. One is located at the wing centerline to provide access to the dry bay over the fuselage. The others are at W.S. 200 and provide access to the nacelle dry bay region. The remaining two doors are located on the lower surface to provide access to the center wing bladder fuel tanks. It is currently planned to have three access doors on the composite center wing, all located on the upper surface. They will be located over dry bay areas, with access to the fuel tanks through the fuel bulkheads.

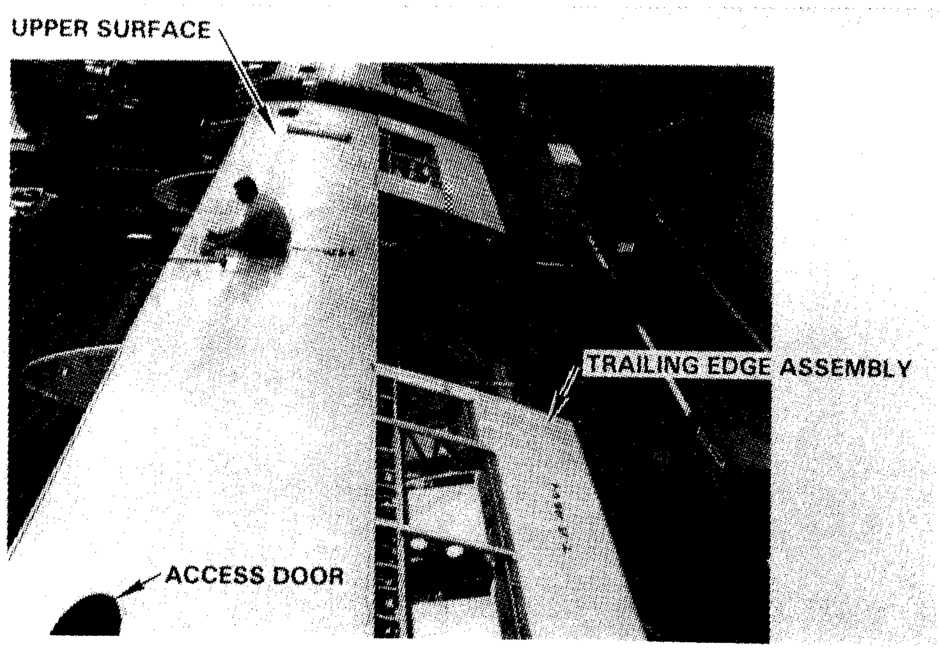


Figure 6

C-130 CENTER WING - W.S. 220 JOINT

Since the composite center wing box must be designed to mate with the existing C-130 metal outer wing, the attachment joint at WS 220 must be compatible with the fastener arrangement on the outer wing. The outer wing to center wing joint is a tension joint with bolts located approximately 6.6 inches apart along the upper and 5.7 inches along the lower wing surface contour. This bolt spacing controls the stringer spacing of the composite center wing box so that adequate structural attachment can be provided.

The C-130 metal outer wing uses a "rainbow"-shaped wing joint fitting at WS 220 that extends from the front to the rear wing beam. A similar fitting is proposed for the upper and lower covers of the composite center wing design that will also extend from the front to the rear wing beam and mate with the "rainbow" fitting on the outer wing.

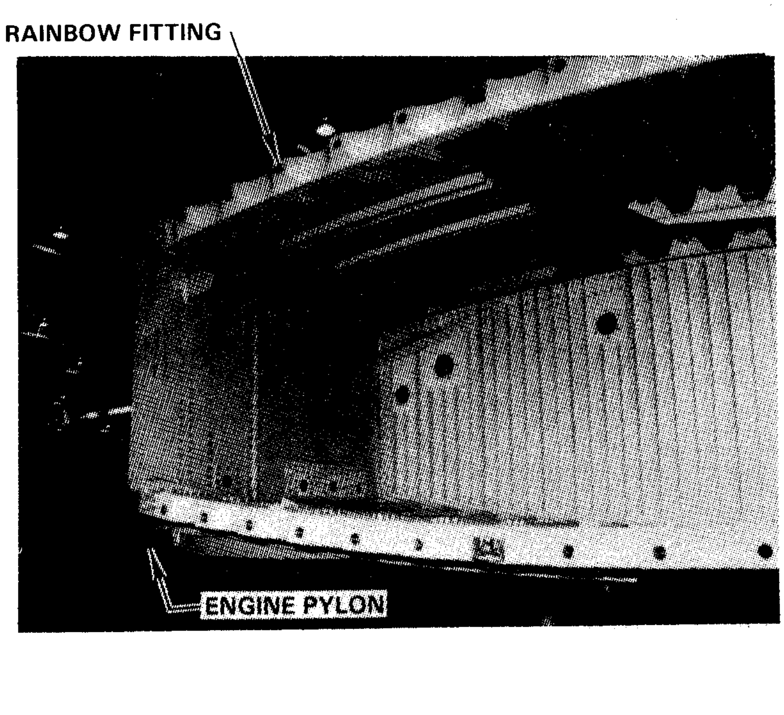


Figure 7

C-130 CENTER WING BOX INTERIOR

An interior view of the metal center wing box is shown in Figure 8. This view is in the dry bay area located over the fuselage in the vicinity of the W.S. 20.5 truss rib. The center wing box consists of ten ribs spaced at approximately 40-inches apart. The ribs include two truss type, four stiffened web configurations and four stiffened fuel tank bulkheads. Since the center wing box cuts across the upper portion of the fuselage, continuity of the fuselage longerons is provided through the box rib structure. Continuity of the Buttline 20 fuselage longeron is provided by attaching to the upper chord of the W.S. 20.5 truss rib. Continuity of the Buttline 61.6 longeron is provided by attaching to the lower chord of the W.S. 61.6 fuel bulkhead rib. This requirement for providing fuselage longeron continuity will be accommodated in the composite design.

Figure 8

TECHNOLOGY ISSUES

This program will develop and demonstrate composite wing design and manufacturing technology to provide the confidence required to commit the design and fabrication of primary aircraft structures to composites. The technology issues which have to be resolved to provide that confidence are shown in Figure 9.

- **DAMAGE TOLERANCE**
- **DURABILITY**
- **LIGHTNING PROTECTION**
- **FUEL CONTAINMENT**
- **HIGH CONCENTRATED LOADS**
- **FABRICATION OF LARGE THICK SKINS**
- **DESIGN OF CUTOUTS**
- **JOINT DESIGN**
- **STRENGTH/STABILITY METHODS OF ANALYSIS**

Figure 9

C-130 CENTER WING-TECHNOLOGY AREAS

The center wing box is of sufficient size and complexity to interrogate fully the many engineering and manufacturing technology issues which must be resolved before a composite wing box structure can be confidently applied to large transport aircraft. Some of the technology issues to be addressed are illustrated in Figure 10. Specific features of the advanced composite center wing make it particularly suitable for addressing these technology issues. These features include:

- Access doors - cutouts must be provided for large access holes in the upper cover.
- Integral fuel tank - the composite wing will incorporate an integral fuel tank system instead of the bladder tank arrangement used on current C-130 aircraft for fuel containment. Tank sealing and lightning protection must be addressed.
- Large structure - develop manufacturing technology for fabrication of large, thick skins economically.
- Pylon attachments, flap attachments and chordwise production joint - efficient joints for transfer of large concentrated loads must be developed.

The use of "toughened" resin systems to increase the design strain level restrictions normally used for damage tolerance control will also be investigated.

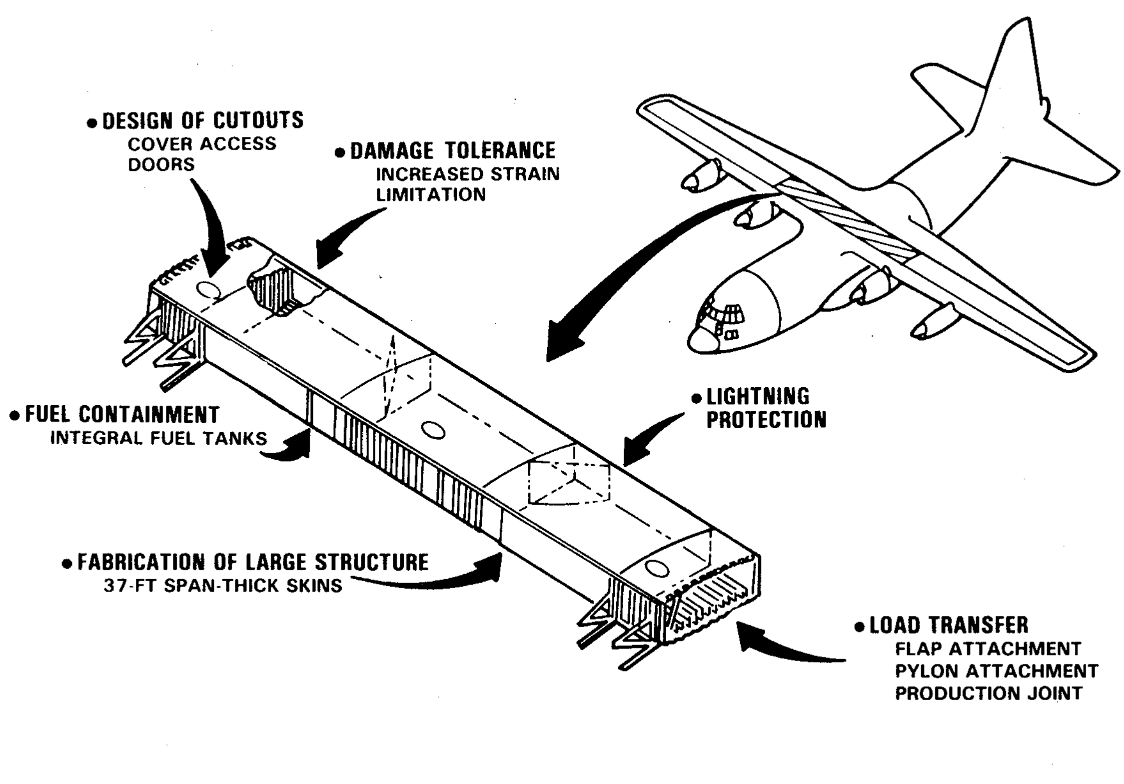


Figure 10

WEIGHT AND COST GOALS

The driving force behind the application of advanced composite materials to aircraft structures is the potential for significant weight savings and reductions in production costs. The application of advanced composites in redesigned secondary and medium primary components of large aircraft in the ACEE programs have demonstrated structural weight savings of 22 to 29 percent over their metal counterparts. Cost reductions were gained by the reduction in the number of parts and fasteners in components designed with composite materials. The cost and weight goals established for the composite center wing program are shown on Figure 11. A weight savings of at least 25 percent will be demonstrated when compared to a replacement state-of-the-art metal wing. The benefits of aircraft resizing will be included in the weight saving assessment. The cost goal established for the composite component is that the acquisition cost in production quantities will be 10 percent less than the cost of a metal wing box designed to meet the same requirements.

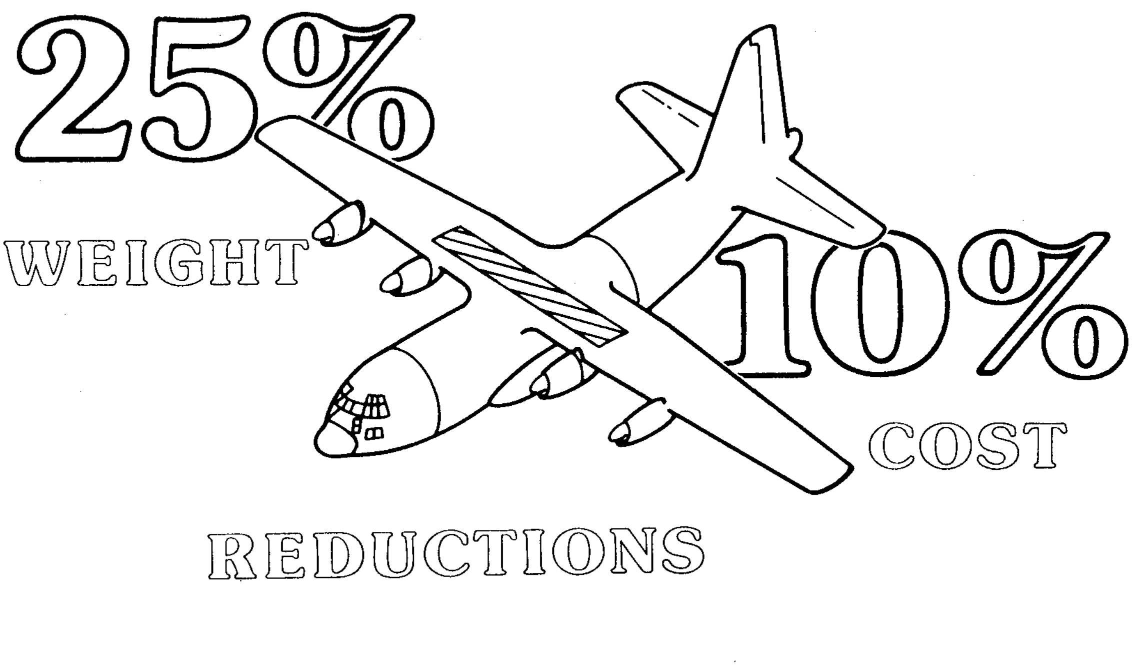


Figure 11

DISTRIBUTION OF WORK

This program is being conducted jointly with the Lockheed-Georgia Company as shown in Figure 12. The Lockheed-California Company has overall program management responsibility for this program and will select the advanced composite material system, develop material specifications and design allowables, and specify the lightning protection system for the advanced composite center wing. In addition, Lockheed-California Company has the responsibility for the design, development, fabrication and testing of the advanced composite covers and ribs. The Lockheed-Georgia Company has overall design responsibility, they will establish structural and interface requirements and will develop a finite element model of the composite box. Lockheed-Georgia will also design, develop, fabricate and test the spars, select fasteners and develop fastener allowables and assemble the wing box. The ground test of the full scale box structure to verify static and damage tolerance characteristics will be conducted by the Lockheed-Georgia Company.

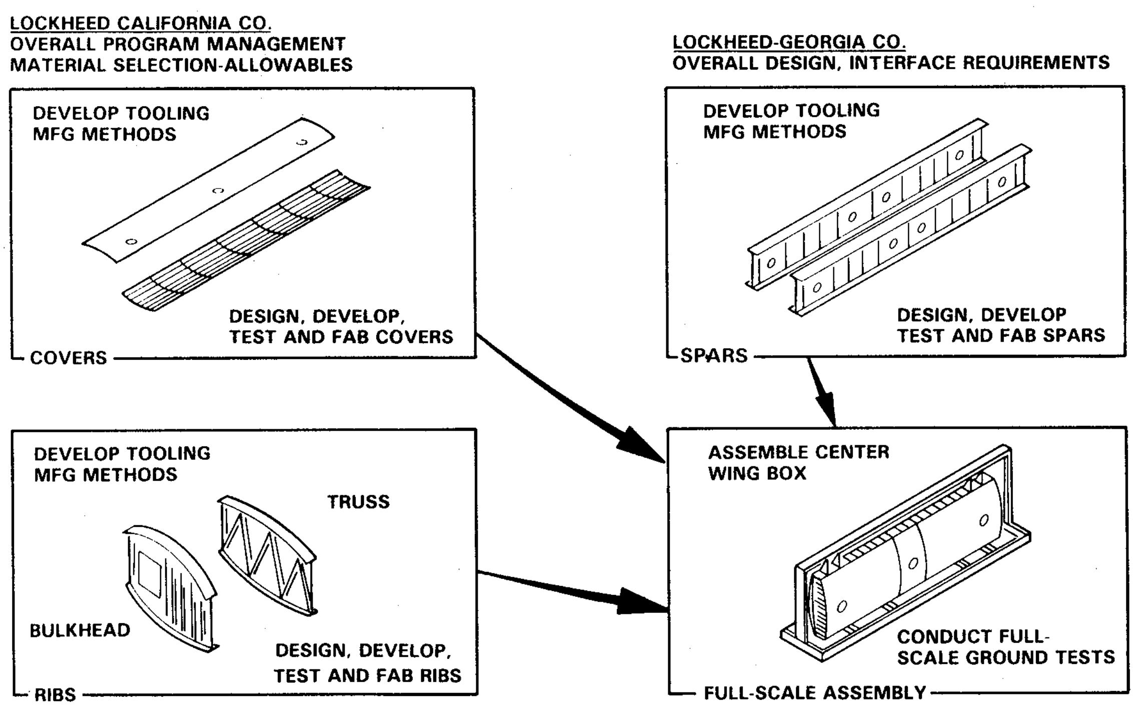


Figure 12

PROGRAM MASTER SCHEDULE

The advanced composite center wing program, illustrated in Figure 13, is organized in four overlapping phases. The interrelation of related contract research and Lockheed's inhouse activities with this program is also shown. Technology and data being developed under NASA's Wing Key Technology programs are being used in this program as well as the wing manufacturing technology being developed under the Air Force's Mantech program.

In Phase 1, Engineering Development, detail design and analysis of the selected composite wing component configurations will be conducted. Subcomponent development tests will be identified and engineering drawings of the full-scale components will be completed. In Phase 2, Design Development Tests, subcomponents representative of the cover, rib and spar configurations will be designed, fabricated and tested for verification of design approaches and fabrication procedures. In Phase 3, Wing Component Fabrication and Assembly, full-scale covers, ribs and spars will be fabricated and assembled for test. During Phase 4, ground tests will be conducted on the full-scale box structure. The tests will include static, durability and damage tolerance.

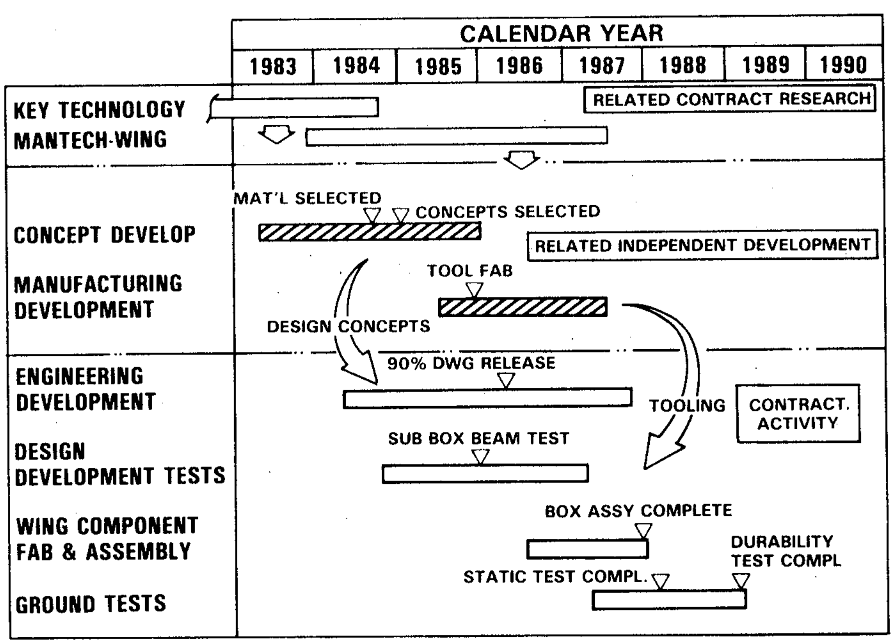


Figure 13

CANDIDATE GRAPHITE/EPOXY MATERIALS

Within the last several years dramatic improvements have been made in the mechanical properties and processing behavior of graphite/epoxy composites. Fiber tensile strengths have been increased by 70 percent and new tougher resins have been introduced. Several of the new graphite/epoxy materials have been investigated as part of the Wing Key Technology Program (Contract NAS1-16856).

Other fiber/resin systems are continually being evaluated as part of various Company-funded programs. Materials are being investigated to determine cure characteristics, chemical properties, mechanical properties, and producibility. When this in-house program is completed the data will be compared to data obtained on other fiber/resin combinations and a selection made for the composite wing box.

Various forms of the graphite/epoxy material, tape, preplied tape, or woven fabric, are also being investigated as part of various Company-funded programs. Design trade studies are being conducted which compare the fabrication costs of the structure as a function of material form and lay-up technique (machine or by hand). The objective of this activity is to determine which material form offers the lowest fabrication costs while meeting structural requirements and weight savings goals.

FIBER	RESIN	RESIN SUPPLIER	PROGRAM INVESTIGATING MATERIAL
AS4	2220-1	HERCULES	NASA KEY TECHNOLOGY
AS4	3502	HERCULES	
CELION	982	AMERICAN CYANAMID	
HS CELION	1504	HEXCEL	
HS CELION	5245C	NARMCO	
HS CELION	1806-2	AMERICAN CYANAMID	IN HOUSE
HS CELION	HST-7	AMERICAL CYANAMID	
AS6	2220-1	HERCULES	
HS CELION	974	FIBERITE	

HS HIGH STRAIN

Figure 14

POST-IMPACT COMPRESSION STRENGTH

Post-impact compression tests were conducted on laminates impacted at 20 ft-lb and 30 ft-lb. A complete description of the test fixture and procedures for this test can be found in NASA Reference Publication 1092, May 1983 "Standard Tests for Toughened Resin Composites."

A comparison of the post-impact compressive failure strain versus impact damage area is presented in Figure 15 for four of the materials being investigated. The resin content of the laminates tested is also presented. The high strain Celion/HST-7 material has the highest post-impact compression strain to failure, however this material also has an exceptionally high resin content, which would decrease its structural efficiency. The AS6/2220-1 material did not perform as well as the other materials which were tested, probably because of the large amount of damage caused by the 20 ft-lb and 30 ft-lb impacts. The failure strains for the AS6/2220-1 material are lower than those reported for AS4/2220-1. Both the high strain Celion/1806 and high strain Celion/974 materials have post-impact compression failure strains which are far greater than untoughened graphite/epoxy composites.

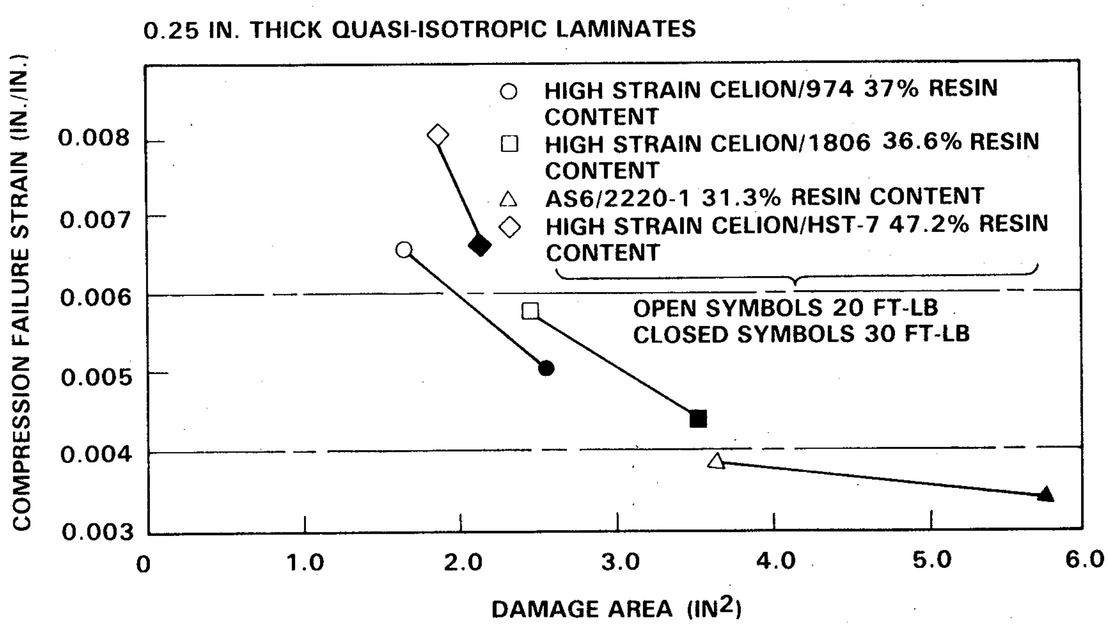


Figure 15

NOTCHED TENSILE STRENGTH

Quasi-isotropic laminates (approximately 0.080 in. thick) were tested in tension in the notched condition. The coupons were 2.0 in. wide by 14.0 in. long and had a 0.25 in. diameter open hole in the center of the coupon. Test procedures are described in the previously referenced NASA Reference Publication 1092.

A comparison of the test data, shown in Figure 16, indicates that all the materials tested demonstrated excellent fiber tensile strength translation. It is anticipated that a tensile design allowable for the notched condition of 6000 μin./in. is easily achievable.

Additional tests are being conducted on each material to ascertain hot-wet compression properties, strain energy release rate, unidirectional laminate tensile and compression properties.

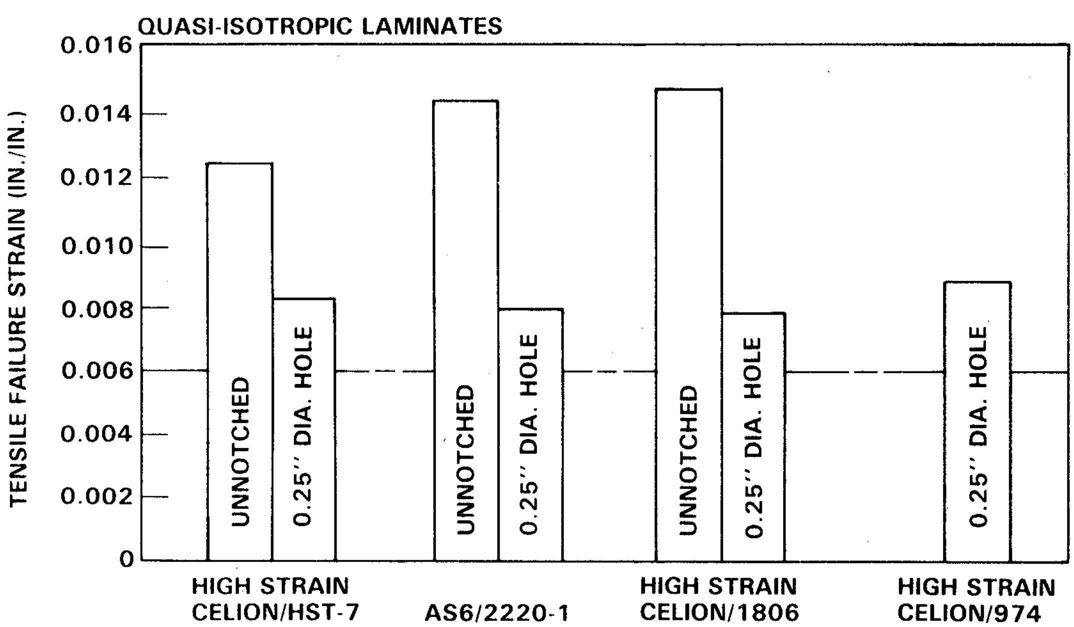


Figure 16

WEIGHT SAVINGS FOR BLADE STIFFENED DESIGN

Up until now, the factors constraining the utilization of the high strength properties offered by advanced composites were: poor matrix toughness, which results in a dramatic reduction in compressive strength due to impact damage, and low fiber elongation, which, in combination with poor matrix ductility, results in a significant reduction in tensile strength of a laminate due to a notch. Advances are being made in the formulation of new resin systems which offer increased toughness without a drastic reduction in hot-wet mechanical properties. Although several tougher systems with increased fiber elongation are available now, as identified on the previous figure, further improvements can be expected within the next few years.

In anticipation of these improvements in graphite/epoxy systems, the effect of design allowables on wing surface weight was evaluated as part of Lockheed's Wing Key Technology (NAS1-16856) program. The results for an optimum blade-stiffened upper and lower wing surfaces are shown on Figure 17. As shown on this figure, the weight saving potential for wing structure designed with a graphite/epoxy composite system which will permit the design strain level restrictions for damage tolerance control to be increased from 4000 μ in/in to 6000 μ in/in is significant.

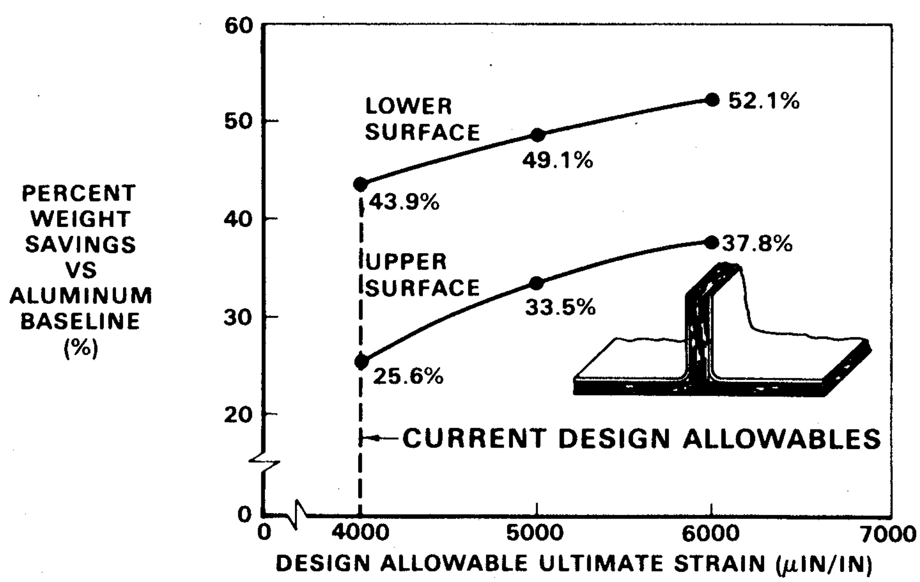


Figure 17

ENGINEERING DEVELOPMENT OF COVERS

The primary design goal for this program is to obtain a weight savings of 25 percent and a cost savings of 10 percent when compared to an aluminum wing box. Therefore, a wide variety of design concepts must be evaluated on each of the wing major subcomponents to determine which concept best meets the design goals.

Some of the challenges influencing the selection of the cover design concept are noted on Figure 18. The wing cover joint provides a challenge in that the stiffening element spacing is influenced by the joint bolt spacing, 6.6-inches on the upper surface and 5.7-inches on the lower surface. The type of stiffeners used will have a significant impact on being able to accommodate fuel sealing requirements. The cutouts will be a major challenge in the design of the panel reinforcement around the cutout. Perhaps the biggest challenge will be in providing a design concept which will be simple to manufacture with low fabrication costs.

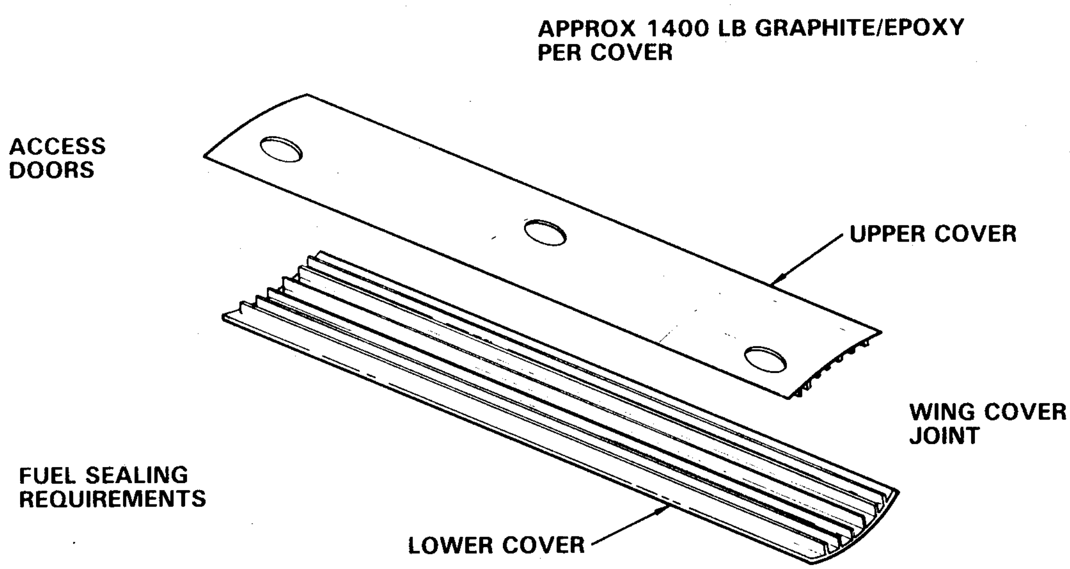


Figure 18

Five design concepts for the covers were selected for study; blade stiffened, "I" stiffened, corrugation stiffened, sandwich, and orthogrid. Geometric variables included the skin and stiffener dimensions and rib spacing. After preliminary sizing each design was evaluated for weight, cost, and a group of qualitative factors such as manufacturing complexity, inspectability, repairability, etc. The results of this evaluation led to the selection of three concepts for further study: blade stiffened, sandwich configuration, and corrugation stiffened.

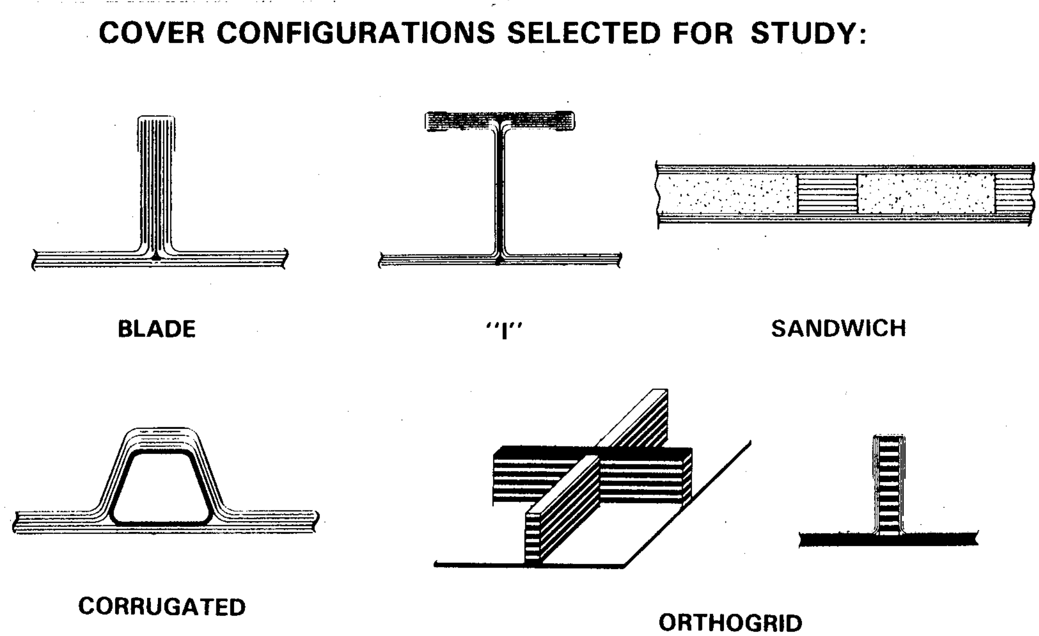


Figure 19

COVER DESIGNS SELECTED FOR FURTHER STUDY

Each of the selected designs are being evaluated for alternate materials, tape versus fabric, and other design refinements. Several of the alternate concepts and materials being studied for each of the selected configurations are shown in Figure 20.

The blade-stiffened concept consists of a pultruded tee cocured with pans to the skin. The height of the blade is tapered to accommodate reduced loads in the outboard sections. The sandwich concept consists of two graphite/epoxy faces separated by a core that consists of a glass/graphite/epoxy hybrid frame with central cavities filled with lightweight polymide foam. Impacted test specimens are being used to evaluate the feasibility of this concept. The corrugation concept consists of another skin and an inner skin providing the corrugation in encapsulating the filament wound trapezoidal insert. Subsequent manufacturing studies has indicated this concept may not lend itself to low cost fabrication.

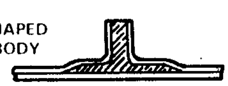
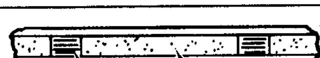
DESIGN	ALTERNATE CONFIGURATIONS	SKIN ORIENTATION	MATERIAL FORMS
BLADE STIFFENED	• SHALLOW TAPER ON BLADE BODY  • 'TEE' SHAPED BLADE BODY 	• "HARD" SKIN 20-50% 0° MATL • "SOFT" SKIN 70-80% ±45° MATL	• TAPE • FABRIC • PULTRUDED BLADE
SANDWICH	• 	• "HARD" SKIN 20-50% 0° MATL	• TAPE • FABRIC
CORRUGATION STIFFENED	• SKIN & CORRUGATION APPROX. SAME THICKNESS  • TRAPEZOIDAL INSERT IS MAIN LOAD CARRYING ELEMENT 	• "HARD" SKIN 20-50% 0° MATL • "SOFTSKIN" 70-80% ±45° MATL	• TAPE • FABRIC • FILAMENT WOUND TRAPEZOIDAL INSERT

Figure 20

ENGINEERING DEVELOPMENT OF RIBS

The C-130 advanced composite box has two types of ribs, web and truss. The current rib arrangements results in four truss type ribs, two fuel tank bulkheads, and four partial truss and fuel bulkhead type ribs as illustrated in Figure 21. The rib spacing is approximately 40-inches. Access to the integral fuel tank areas is provided through access panels located in the fuel bulkhead webs where required.

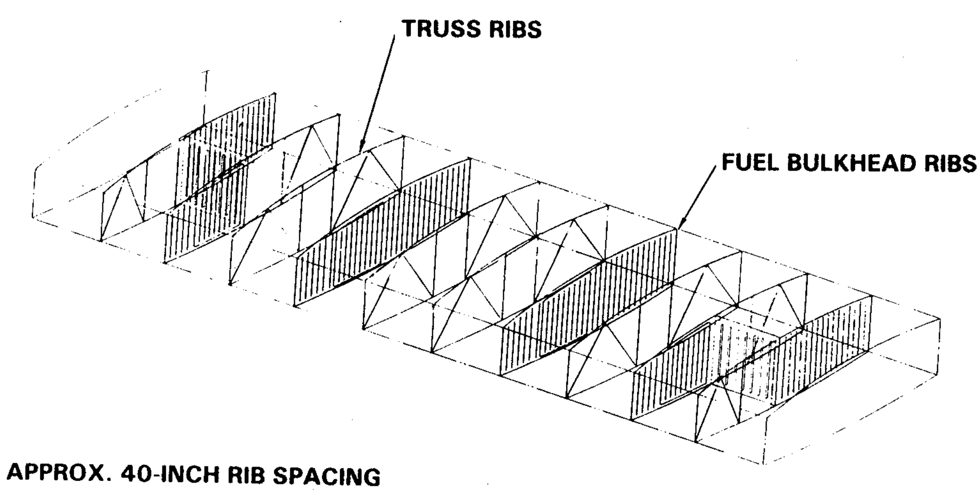


Figure 21

DESIGN/MANUFACTURING TRADE STUDIES-RIBS

Design trade studies, similar to those being conducted on the covers, are also being done for the ribs. As with the covers, design variables will include configurations and materials. The rib cap design is strongly affected by the selected cover concept. If the blade-stiffened cover concept is the final selection, "mouse holes" in the rib cap are required. Figure 22 shows some of the concepts being studied for the fuel bulkhead webs. Integrally stiffened bulkhead webs were evaluated using either graphite/epoxy tape or fabric. A blade-stiffened bulkhead web using graphite fabric proved to be the most efficient design concept. For the truss ribs, the truss member cross sections being evaluated include; channel, cruciform, square tube and "H" using aluminum and graphite/epoxy.

STIFFENED WEB CONCEPTS SELECTED FOR STUDY:

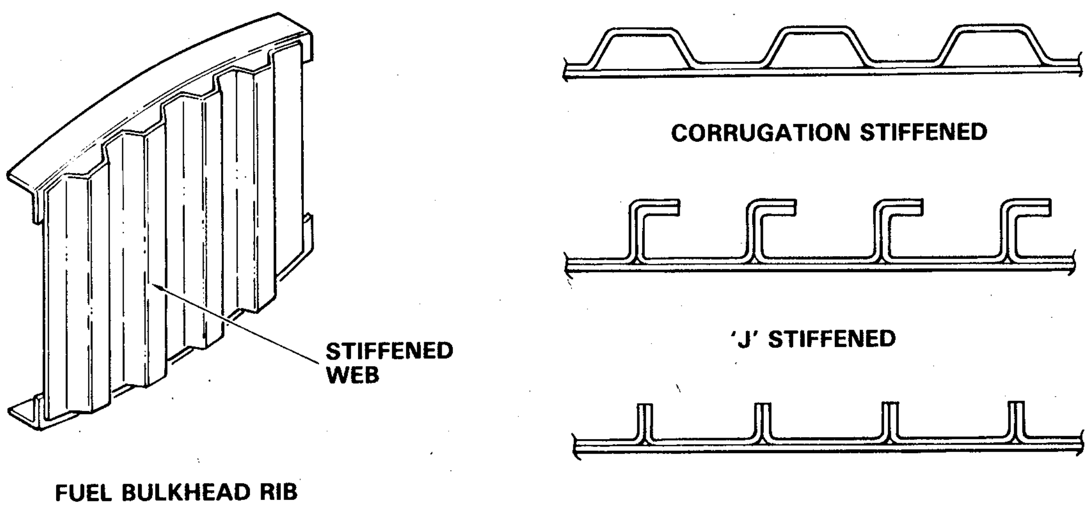


Figure 22

SUMMARY OF COVER DESIGN DEVELOPMENT TESTS

The purpose of the design development tests is to verify the structural integrity of the various wing box subcomponent concepts. Each specimen has been carefully selected to interrogate the integrity of an important design detail of the structure. Specimen design will represent final detail design. The cover design development tests are illustrated in Figure 23.

All test specimens will be fabricated in a production environment using verified manufacturing processes. All test specimens will be subjected to the inspection procedures required for full-size components. These include material receiving, in-process control, cure cycle verification, visual and dimensional inspections, process control tag-end tests, and nondestructive inspection.

In addition to the cover subcomponent tests, two large box beam specimens will be fabricated and tested to integrate the covers, ribs and spars into one component to verify the general structure as well as the interaction of the subcomponents under representative loading. Of particular significance for the covers is the evaluation of the access door cutout located in the upper cover of the box beam structure.

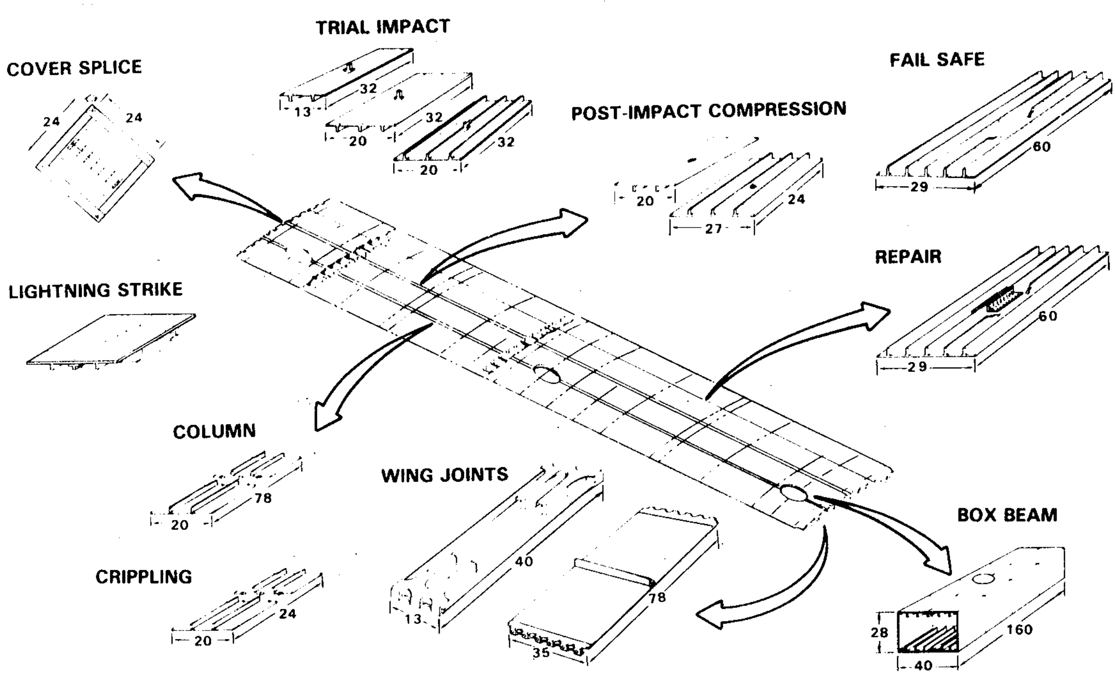


Figure 23

SUMMARY OF RIB DESIGN DEVELOPMENT TESTS

A variety of rib cap structural details and two rib configurations will be fabricated and tested for the design development test program as shown on Figure 24. The rib test specimens were selected to fully interrogate particular aspects of the rib designs in attachment to the covers, transfer of fuselage longeron loads and engine mount fitting loads and fuel pressure loads. Data obtained from these tests will be compared to that predicted analytically for verification of analysis predictions.

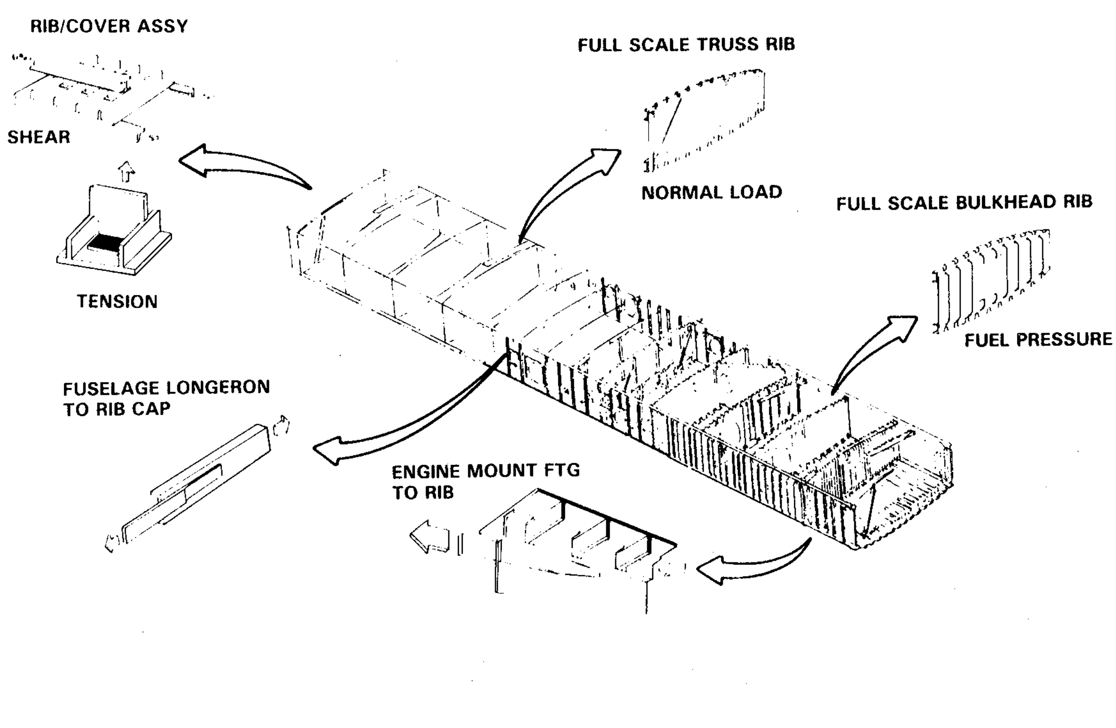


Figure 24

COVER AND RIB FABRICATION AND ASSEMBLY

The Lockheed-California Company has the responsibility for fabrication of the wing covers and ribs. At the current time, one full-scale center wing box is scheduled for fabrication. Figure 25 illustrates the flow of manufacturing activities for the covers and ribs. For planning purposes it was assumed that the cover would be fabricated as two panels, then fastened together as one cover assembly. The cover assemblies would include the rib caps and the rainbow fittings for the attachment of the outer wings. The rib diagonals and fuel bulkhead ribs could be fabricated and shipped with the cover assemblies to the Lockheed-Georgia Company for assembly of the center wing box structure.

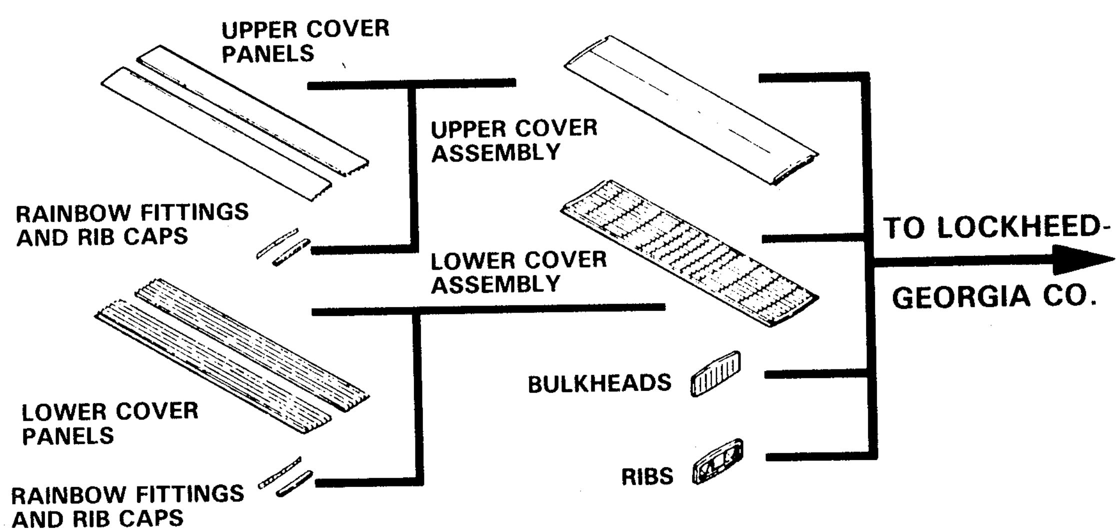


Figure 25

COMPOSITE TRANSPORT WING TECHNOLOGY DEVELOPMENT
PROGRAM REQUIREMENTS - SPAR/ASSEMBLY CONCEPTS

W.E. Harvill, Jr.
Lockheed-Georgia Company
Marietta, Georgia

ACEE Composite Structures Technology Conference
Third Special Oral Review
August 13-16, 1984

INTERFACE REQUIREMENTS

All spar, rib and cover details are designed to be compatible with existing outer wings, engines, fuselage, leading edge and trailing edge assemblies. These interface requirements control, to a large extent, the position of internal structural members; therefore, the location of spars and most ribs for the composite wing remain unchanged. These imposed constraints allow the option for the composite wing to be fitted to an existing C-130H aircraft for possible flight demonstration or to a C-130 derivative for future production application.

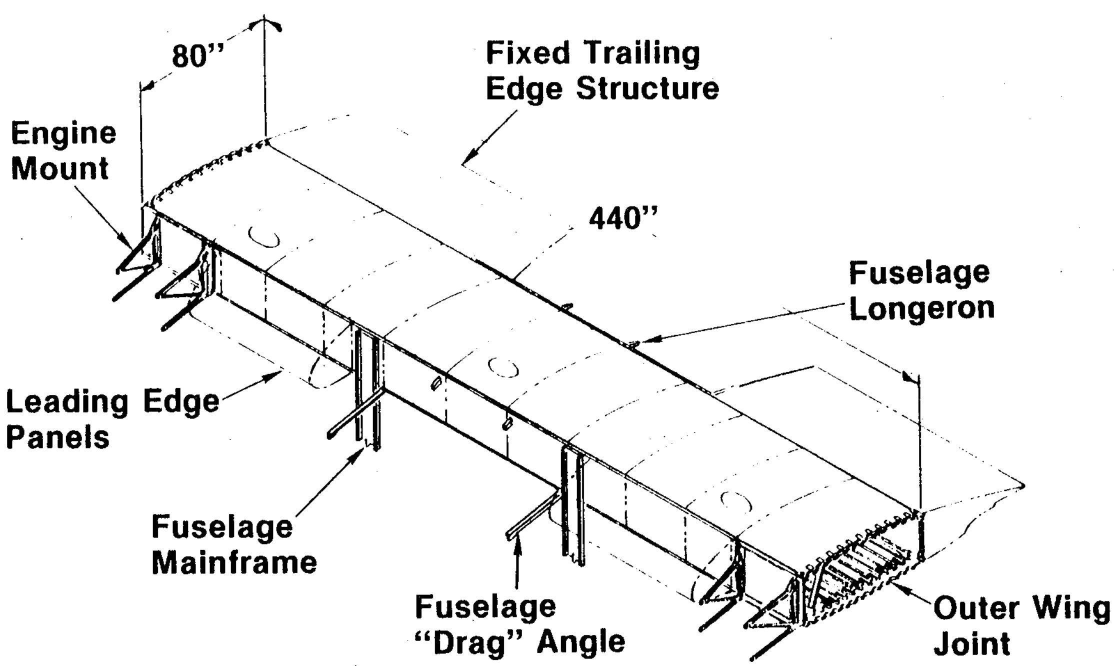


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Figure 1

CENTER WING BOX WEIGHTS - POUNDS

Preliminary design studies show that the weight saving from maximum use of graphite-epoxy in the center wing box of an Advanced C-130 aircraft is about 26 percent. This chart shows the predicted weights of both a metallic and a composite center wing box, designed to the expected structural requirements for a next-generation military cargo transport. As compared to the current C-130 center wing, the metallic Advanced C-130 version would be about 24 percent heavier because of more severe mission requirements; the composite wing box would be about 10 percent lighter. The chart also shows a breakdown of total weight by major components.

Model / Item	Current C-130	Advanced C-130	
		Metallic	Composite
Covers	3155	3999	2882
Spars	758	956	717
Ribs/Bulkheads	516	542	390
Fittings / Misc	382	467	432
Total	4811	5964	4421

$\underbrace{5964 \quad 4421}_{\text{26\% Saving}}$

Figure 2

PROGRAM GOALS EVALUATION

Cost/weight performance is the basis for evaluating the achievement of program goals, and both factors will be closely tracked throughout the program. The accumulated experience is then used in conjunction with expected mission requirements to define two viable aircraft for the 1990's:

- o A military cargo transport, and
- o A commercial transport

These requirements are used with existing preliminary design methodology and selected optimum materials usage to resize the defined aircraft and to provide weight and cost data for comparison with program goals.

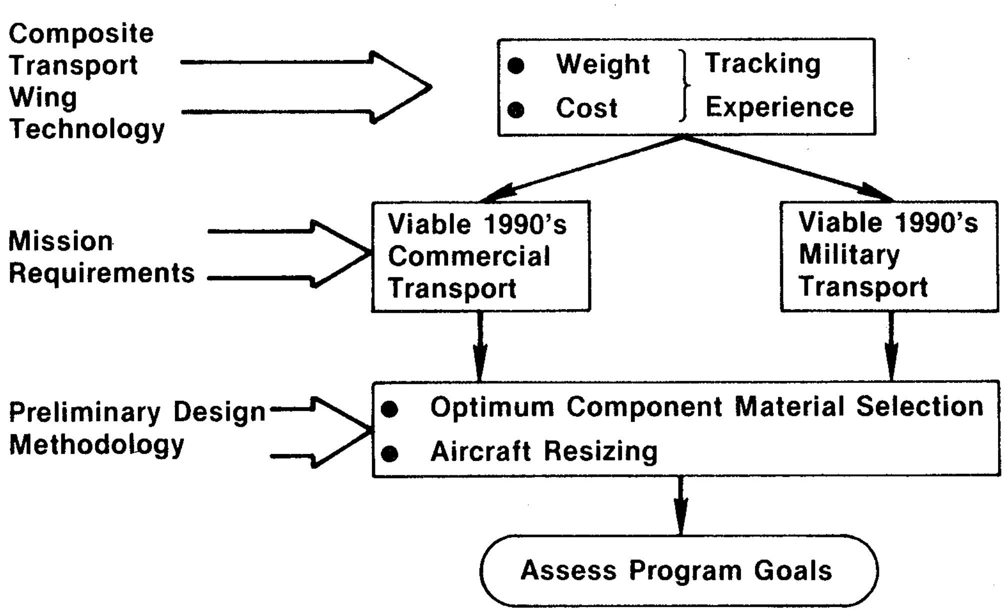


Figure 3

DESIGN REQUIREMENTS

Emerging requirements for the "next generation" military cargo transport could require a higher payload - load factor combination than that currently available in transport aircraft. Sizable improvements in STOL capability are expected - thus more efficient flaps, and landing gear. Improved flight maneuverability at low speed, as a concomitant to STOL, requires better lateral, longitudinal, and roll control. More severe usage requirements, including extended durability and damage tolerance capability, can probably be expected.

The Advanced C-130 is designed to meet these requirements.

- **Increased Payload**
- **3.0g Limit Load**
- **STOL Capability**
  - **New Flaps**
  - **Uprated Engines**
  - **High Sink Rate Landing**
- **Severe Usage Spectra**
- **High Roll Rate**

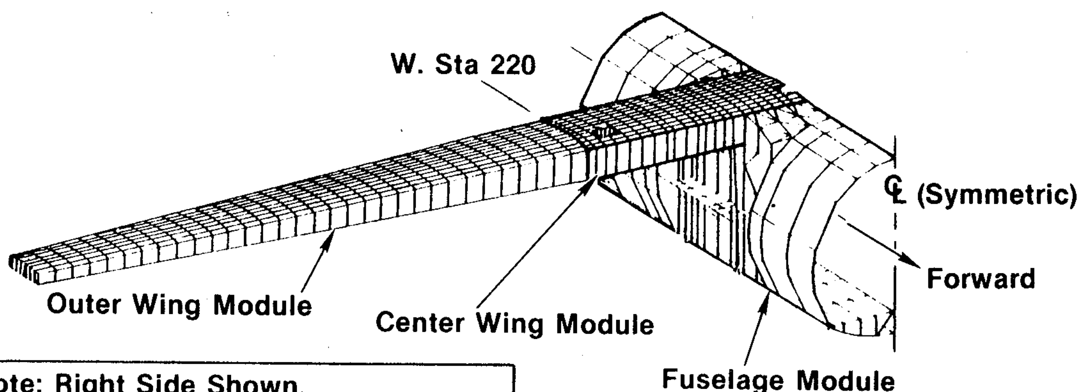
Figure 4

FINITE ELEMENT MODEL

A NASTRAN finite element model was used to generate internal loading of the center wing structure from the application of 50 external load design cases. The model consists of three substructures coupled together to represent the entire wing and the fuselage center section.

The model was analyzed for design cases and fuselage pressurization using COSMIC NASTRAN level 17.6 with automated substructure coupling. In addition to this overall wing model, detail NASTRAN models will be generated for selected areas to further interrogate the center wing box structure. These detail models will use boundary conditions from the presented overall wing model.

NASTRAN MODEL REPRESENTS MAJOR STRUCTURAL MEMBERS



Note: Right Side Shown.
Left Side Obtained
From Symmetric Transformation
During Coupling.

Figure 5

The following figure presents the ultimate center wing cover running load (N_Y) as a function of span. The plotted load is an envelope from maximum upbending and downbending external load cases. It is the total axial load in the skin panels and stringers averaged over the width of the wing box. These loads were calculated using the detailed wing-fuselage NASTRAN model (previously described). For comparison, this loading represents a 24 percent increase over the C-130H ultimate cover N_Y. The maximum N_Y loading, in excess of 25,000 lbs/in., represents a relatively high design load level for current large transport aircraft.

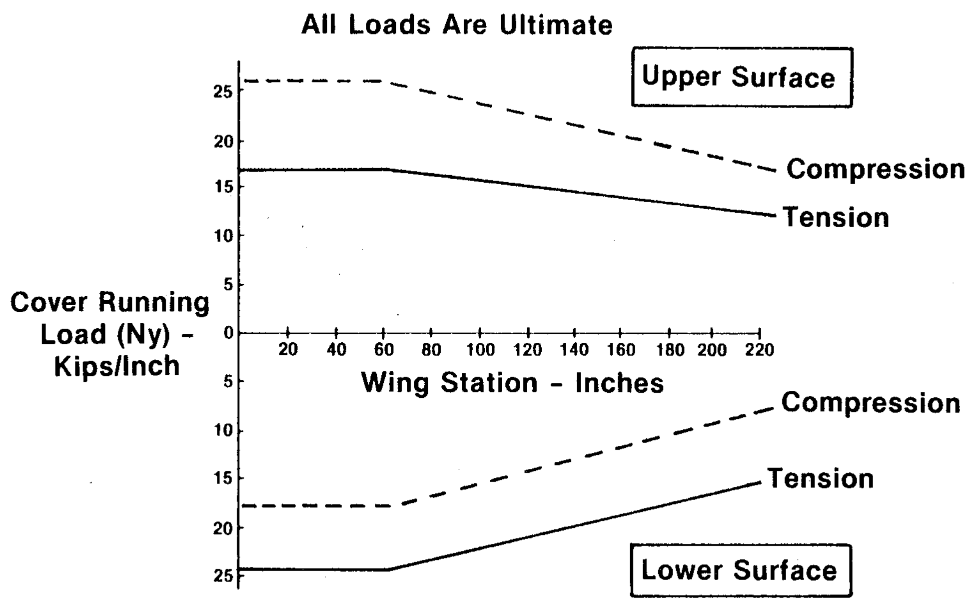


Figure 6

Ultimate envelope spar shear flow is presented here as a function of the center wing span. The maximum spar shear flow from the wing mainframe reaction (W.S. 61) outboard to the center wing-to-outer wing joint results from symmetric external load conditions. The shear flow between the left and right mainframes (W.S. 61L to 61R) reults from unsymmetrical external load conditions.

These shear flows represent a 28 percent increase over the ultimate shear flows currently experienced on the C-130H aircraft.

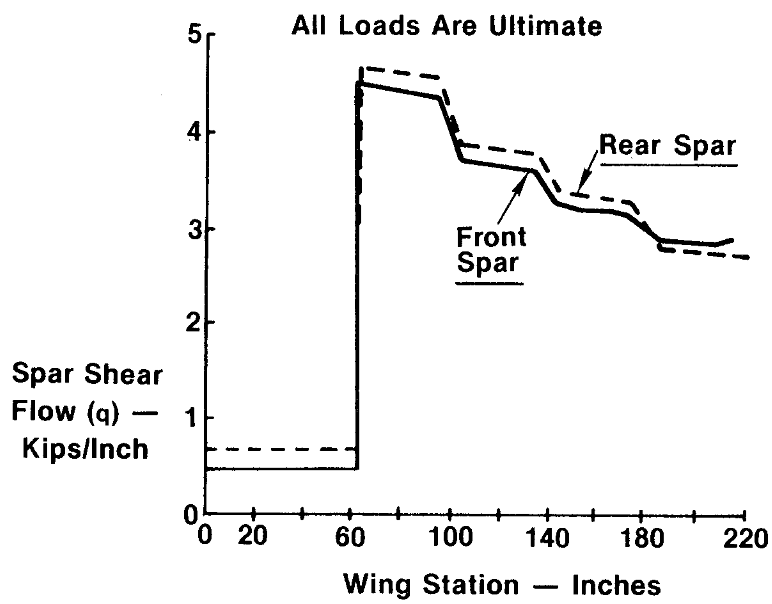


Figure 7

FRONT SPAR NASTRAN CAP LOADS - ULTIMATE

Typical ultimate spar cap forces are shown as a function of span, for the front spar. The cap loads are calculated envelope values from a detailed wing and fuselage NASTRAN model. The loads are applied to a composite cap made up of the spar flange, part of the web, and an appropriate width of cover.

The upbending loads were defined by symmetric maneuver conditions. The downbending envelope was defined by Landing/Ground Handling conditions. Rear spar cap loads are similar.

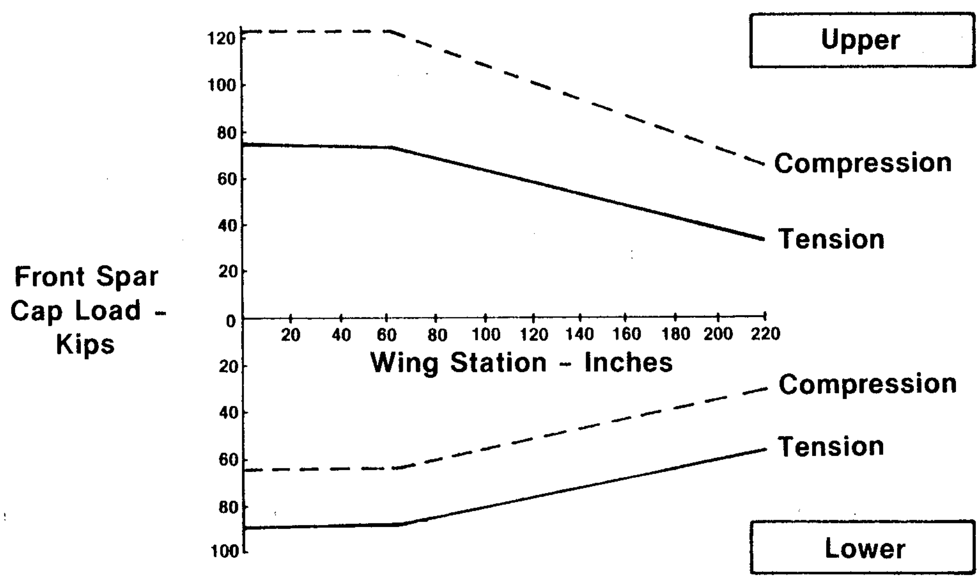


Figure 8

DISTRIBUTION OF WORK

The work to be performed at Lockheed-Georgia is highlighted. In addition to overall design and interface responsibility, the Georgia Company will define joint allowables, select the design configuration and manufacturing processes for front and rear spars, and provide fabrication/test verification for each. The final assembly will incorporate these spars with covers and ribs from Lockheed-California into a complete box for extensive tests at Lockheed-Georgia.

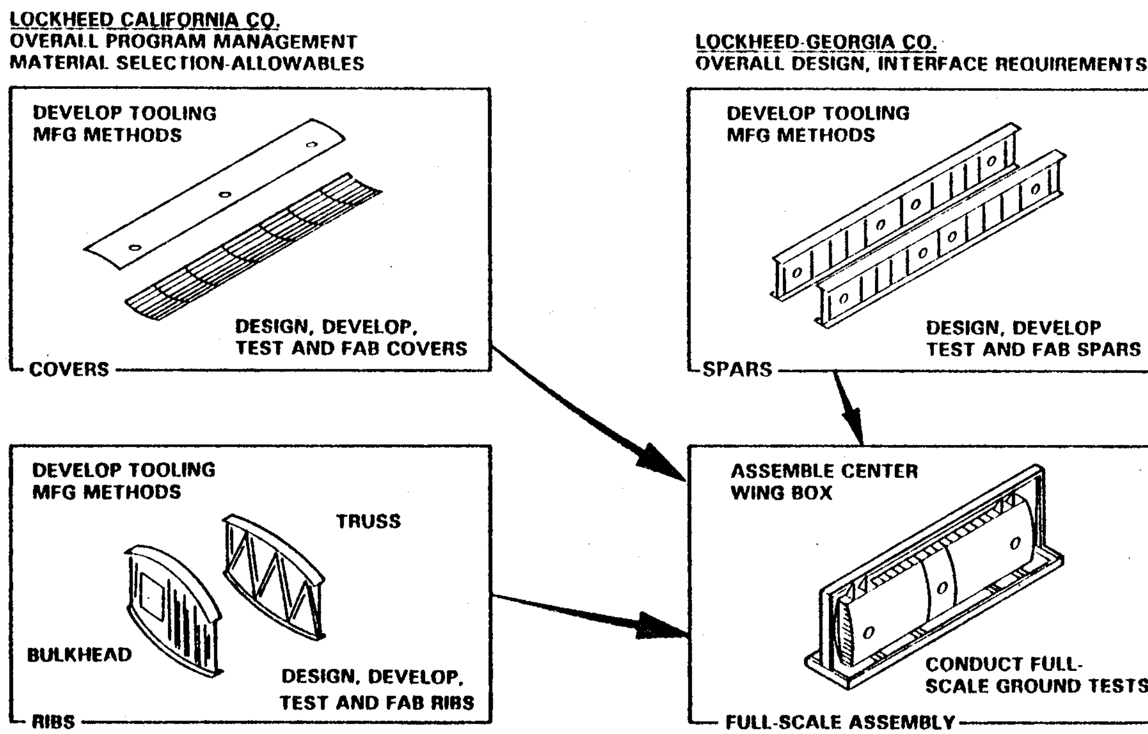


Figure 9

ONE-PIECE INTEGRALLY MOLDED DESIGN

In selecting the design configuration, a number of spar concepts are being evaluated through formal trade studies. Each of these emphasizes reduction in the number of detail parts and fasteners required. The effects of alternate materials, material forms, fastening methods, tooling and fabrication techniques are also being evaluated.

One spar configuration evaluated is similar to that successfully demonstrated on the L-1011 Advanced Composite Vertical Fin Program, but the spar is much longer and accommodates much higher load requirements. When compared to the C-130 metal center wing front spar this single component replaces approximately sixty aluminum details and eliminates the need for over 4000 fasteners.

The spar cap and web thicknesses shown are for the maximum loading conditions and can be reduced by dropping plies in less critical areas.

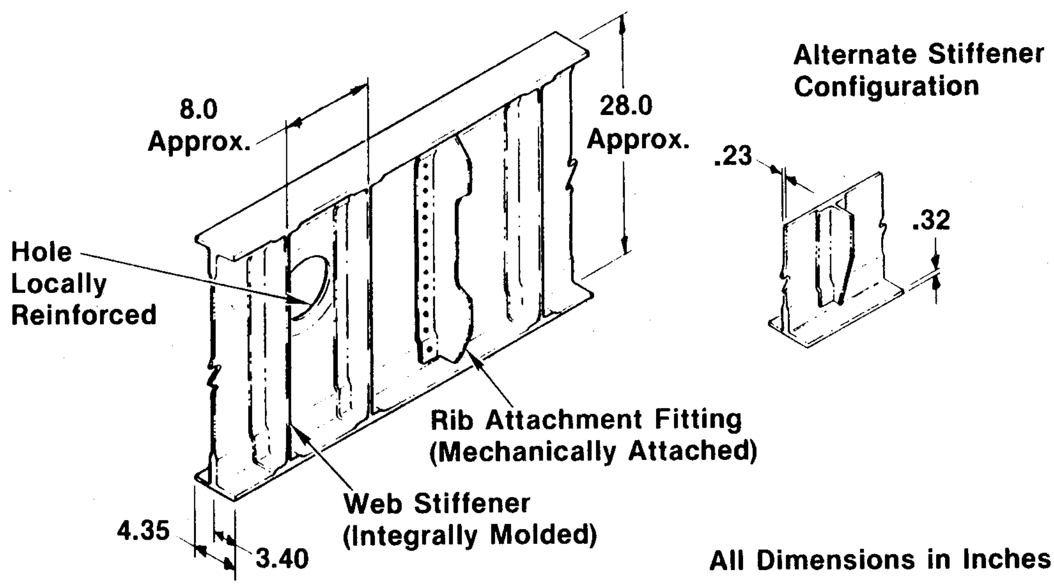


Figure 10

HONEYCOMB WEB DESIGN

The use of non-metallic honeycomb core for spar web stiffening is being evaluated for potential weight and cost advantages. By eliminating the need for separate stiffeners, the time for graphite/epoxy layup is reduced and tooling concepts are simplified.

Preliminary web sizing indicates about 0.1 inch face sheets and 0.38 inch core, with solid graphite/epoxy caps. Potting, pan down, or replacement of core with graphite might be used at attachment locations.

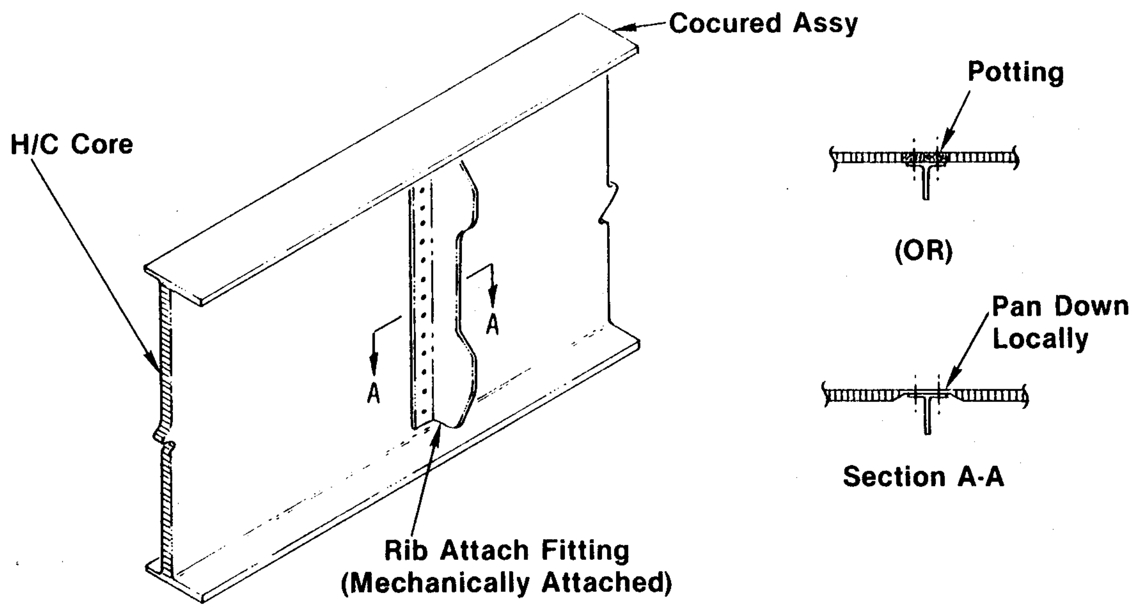


Figure 11

STIFFENED CHANNEL CONCEPT
(FILAMENT WOUND)

The simple channel configuration of this spar concept makes it a candidate for filament winding fabrication methods. By winding directly onto a solid mandrel, spars may be laid up in pairs and subsequently separated following the curing process. Web stiffeners fabricated from pultrusions are also considered for this concept; therefore, the spar assembly maximizes use of automated fabrication techniques.

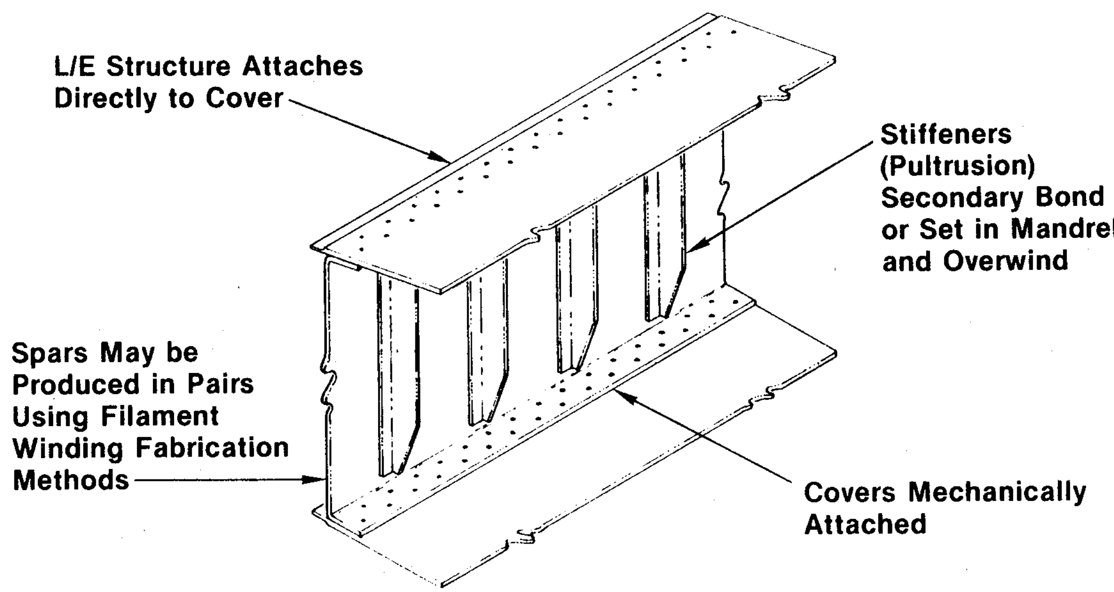


Figure 12

Sine wave spars have been used successfully on several composite wing programs for fighter aircraft. This concept is being evaluated to determine if similar benefits can be anticipated for large transport wings.

Some special design considerations associated with using sine wave webs on the C-130 spar will be encountered in those areas where large cutouts exist and where attachment to wing box and edge structure is required.

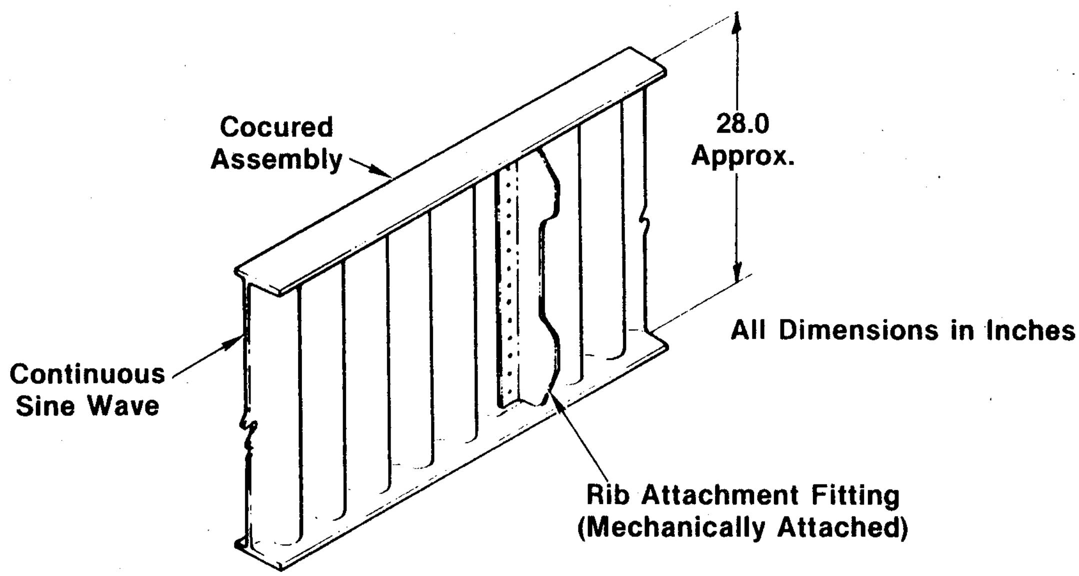


Figure 13

STIFFENED WEB DESIGN

The predominant feature of this spar concept is that spar caps have been incorporated into the covers, producing a simple stiffened web spar amenable to low cost tooling and fabrication methods. If used in conjunction with a blade stiffened cover design, the addition of spar cap elements should not significantly affect the complexity of cover tooling concepts.

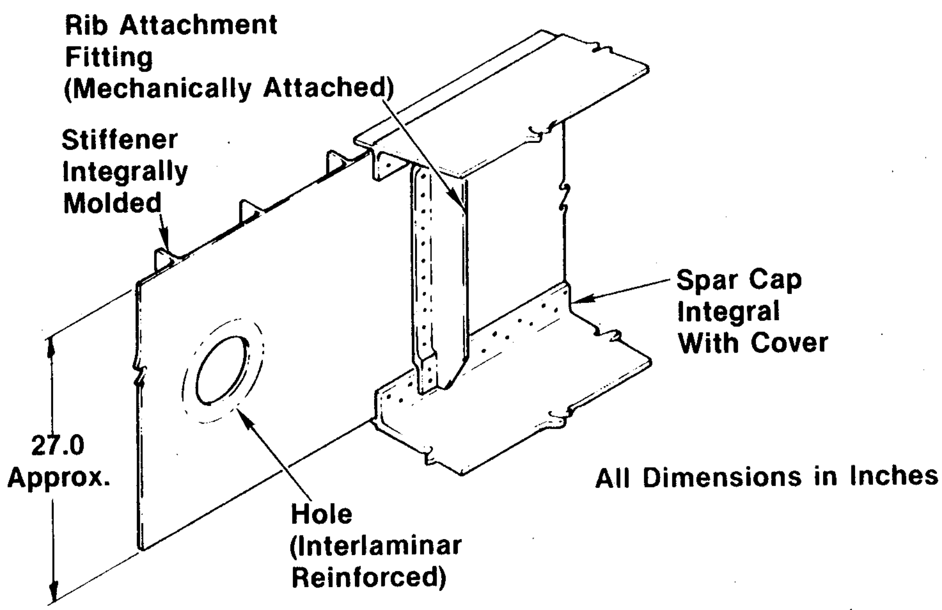


Figure 14

SPAR DESIGN DEVELOPMENT TESTS

Spar design development test specimens represent critical aspects of the spar design in spar-to-cap interfaces, spar webs, and major structural joints.

All test specimens will have sufficient structural detail to adequately test the interaction of loading between the composite elements making up each test component. Test loading will be representative of the critical condition for each test component, and test fixture reactions will be such that realistic internal loading is maintained. Static loading to failure will be performed, as well as durability and damage tolerance testing using flight-by-flight spectra loading.

Data obtained from these development tests will be used in a correlation analysis to compare test specimen response to analytical predictions. Analysis results will be used in the ongoing design effort of the full-scale wing box. This interaction with the design will assure the successful design, manufacture and testing of the full-scale test article.

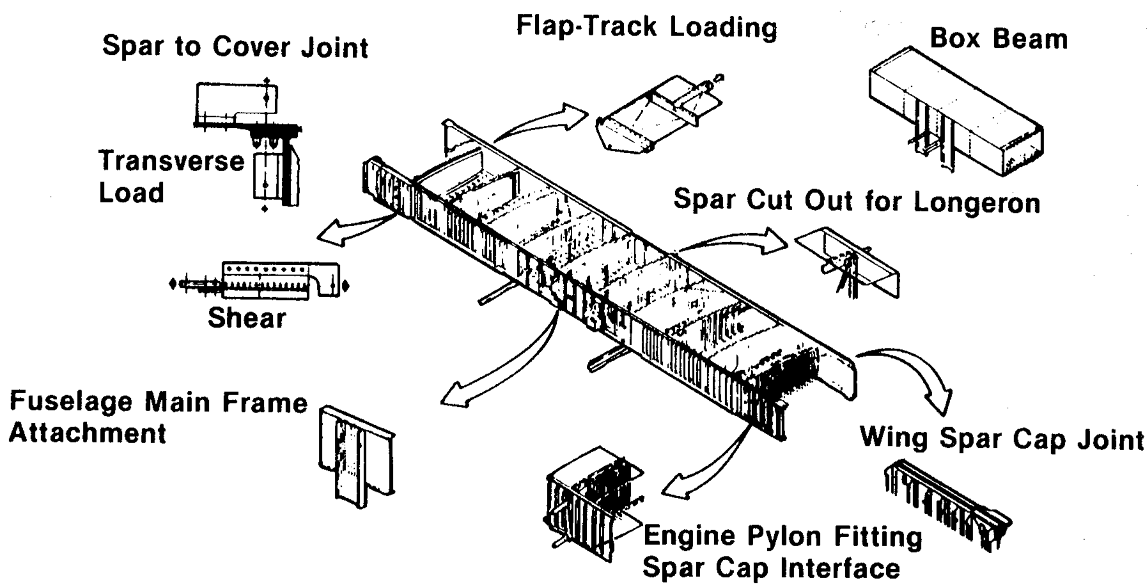


Figure 15

BOX BEAM TEST COMPONENT

The Box Beam Test specimen shown is a reduced size structure containing full-scale features of the center wing. The specimen has full depth spars and full-scale covers representing 50 percent of the wing chord. Major structural joints and representative wing reactions are included, as well as other specific details such as access openings in the cover.

Two specimens will be tested for final verification of the various cover, spar and rib components prior to final design release. These test articles will demonstrate the effects of an internal interacting structure under combined shear, bending and torsion. Multipurpose tests will address fuel containment, lightning strike, static strength, durability, and damage tolerance. A final static test of the durability and damage tolerance specimen will determine the residual strength under a selected critical load condition.

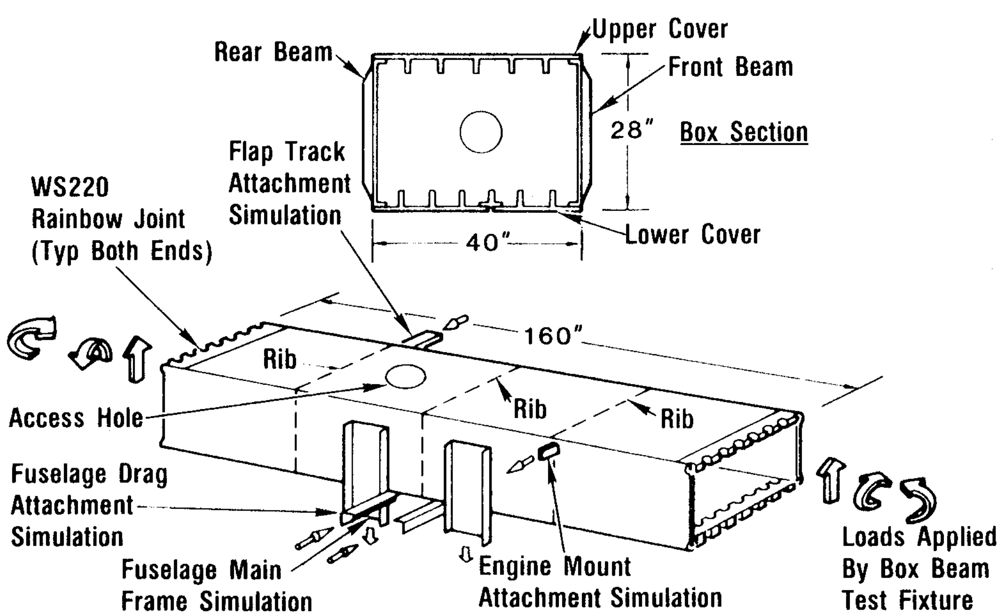


Figure 16

SPAR SIZE

This illustration depicts the size of the spars and the layup fixture as compared to the size of a person. The spars are 440" long and 28" deep, with cap flanges about 4" wide.

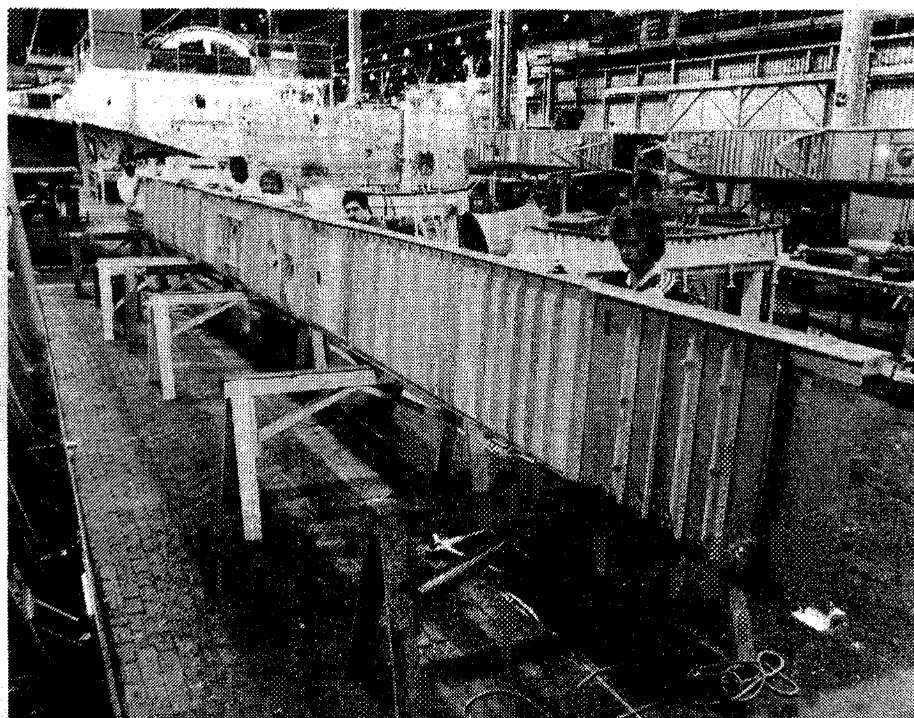


Figure 17

SPAR MOLDING TOOL EVALUATION

One tooling concept being considered incorporates the use of aluminum, graphite-epoxy, and rubber materials. Graphite-epoxy details, next to the spar, aid in the control of thermal expansion. Aluminum is used for the outer base plate and side rails to maintain stability. Two sets of angled side rails accommodate the layup of both the front spar and rear spar on the same tool base. Two molded rubber mandrels are open to autoclave pressure, providing the required pressure against the lower flange and absorbing the expansion of the aluminum plate trapped under the spar web.

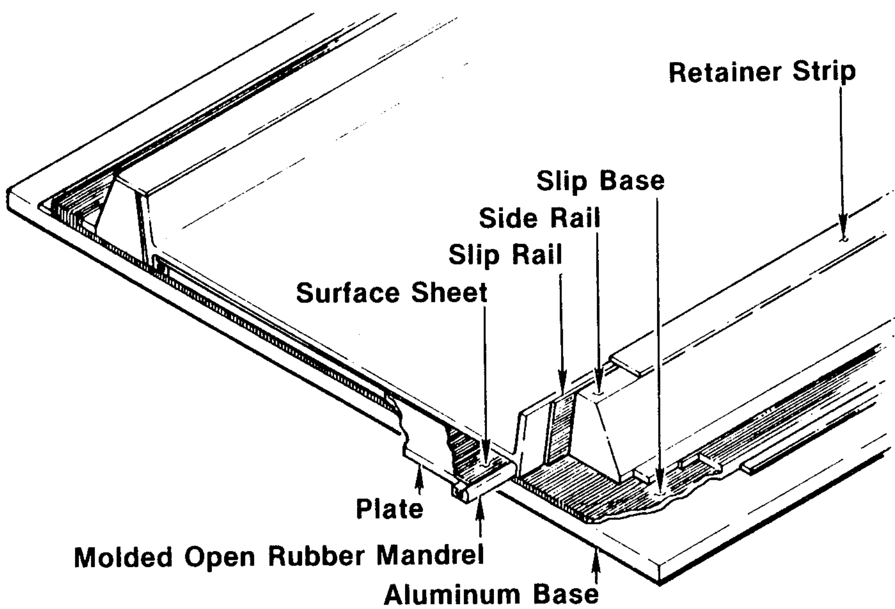


Figure 18

ASSEMBLY SEQUENCE

The box assembly sequence is illustrated. Upper and lower cover subassemblies (including rainbow fittings for the W.S. 220 joint) and ribs are provided by Lockheed-California. Lockheed-Georgia will furnish spar subassemblies, W.S. 220 joint corner fittings, and other hardware. A modified existing production assembly fixture will locate and hold the subassemblies and other parts in place for final mechanical assembly. Titanium fasteners, wet-installed, will minimize corrosion. The completed box will be subjected to a thorough inspection before release for test.

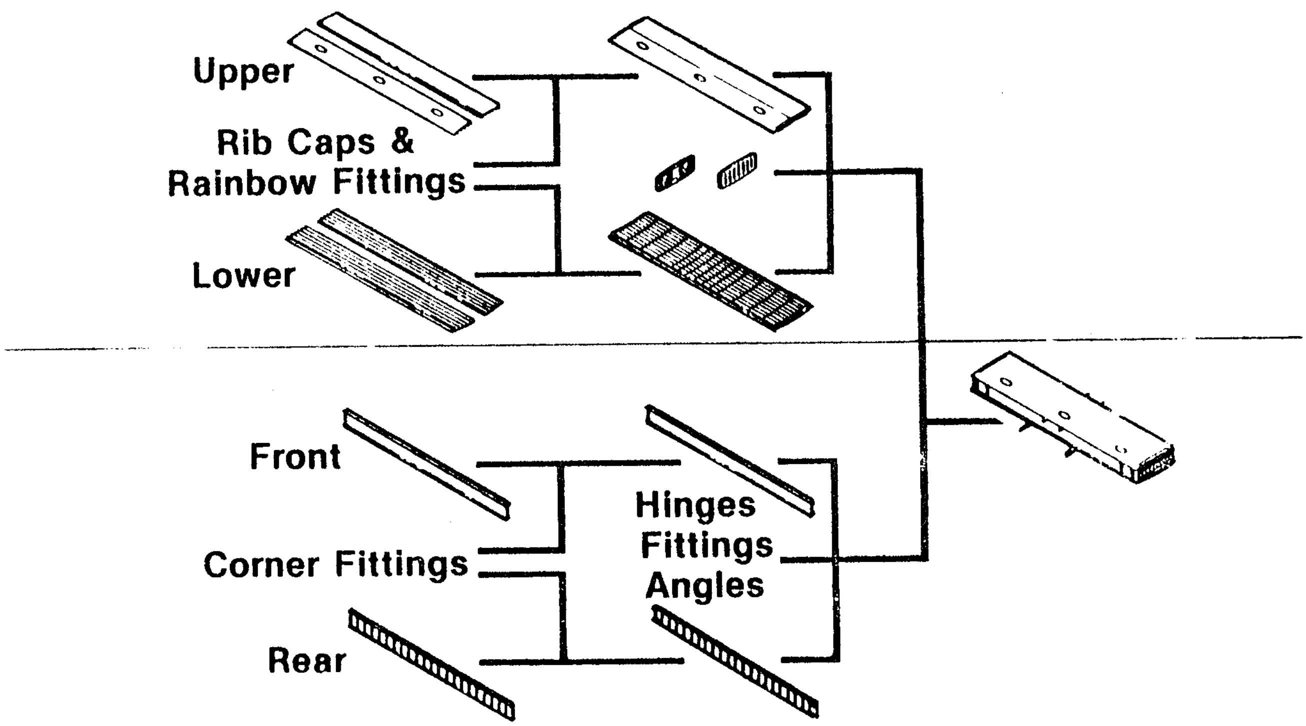


Figure 19

ASSEMBLY CONCEPT "A"

The Advanced Composite Center Wing will be assembled in an existing C-130 assembly fixture suitably modified to accept the various composite details. Major components of the wing box structure will be fabricated separately and subsequently assembled with mechanical fasteners or fastener/adhesive combinations.

A number of assembly concepts are under consideration, but final choice depends on the component configurations selected. As an example, however, the integrally molded rib cap arrangement shown in this concept forms an efficient fuel barrier and should be most beneficial at fuel bulkhead locations. Titanium fasteners will be used exclusively for the assembly of the wing box. The entire internal structure is accessible during assembly and no blind fasteners will be needed.

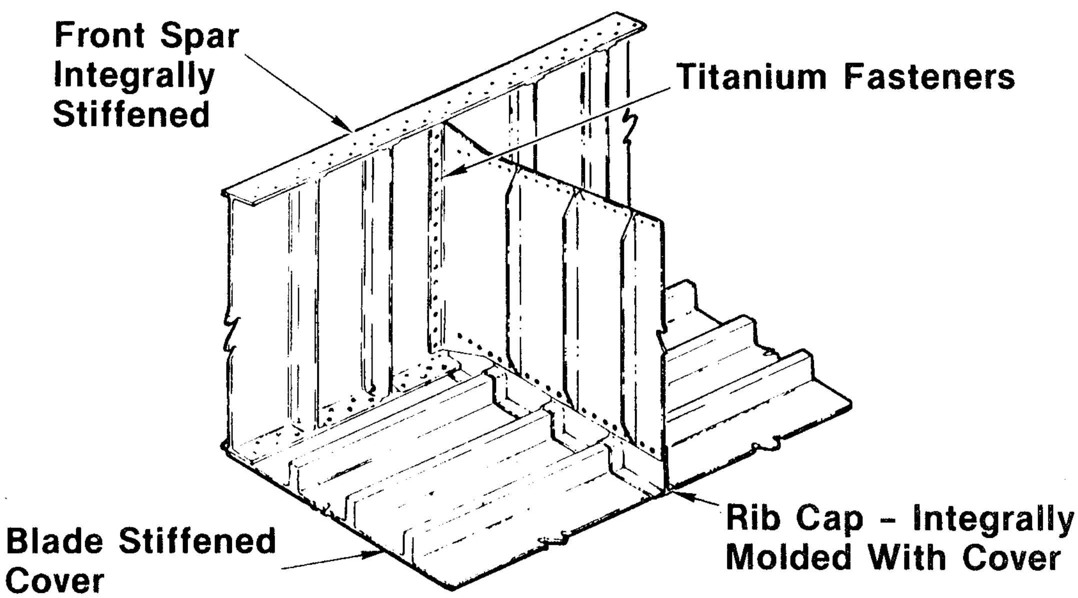


Figure 20

ASSEMBLY CONCEPT "B"

In this concept, the cover attachment fasteners used with the one-piece rib design shown here should act as an efficient barrier in the event of a cover delamination. Potential fit-up problems could be experienced with this concept and shimming could be expected. The use of female tooling, however, for the fabrication of ribs, should minimize this problem. Adhesive bonding is being evaluated as a means of fuel tank sealing and to determine potential structural benefits.

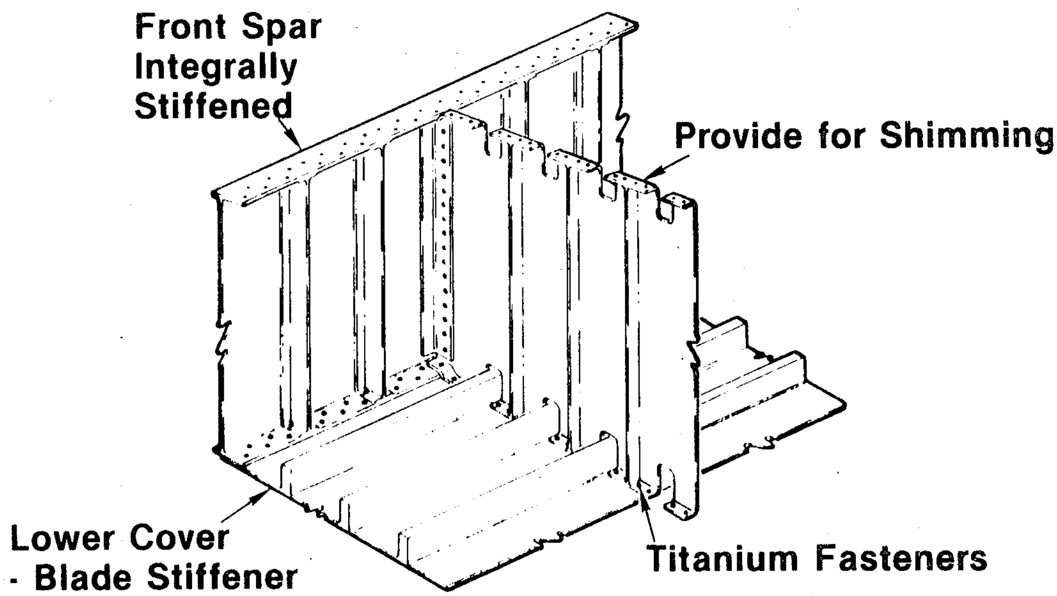


Figure 21

MODIFIED CENTER WING ASSEMBLY FIXTURE

An existing C-130 floor assembly fixture, shown here in a cutaway sketch, will be used to assemble the composite center wing box. The fixture will be modified with removable details to provide for the composite configuration. After the box assembly is complete, the fixture will be restored to provide for the original metallic configuration.

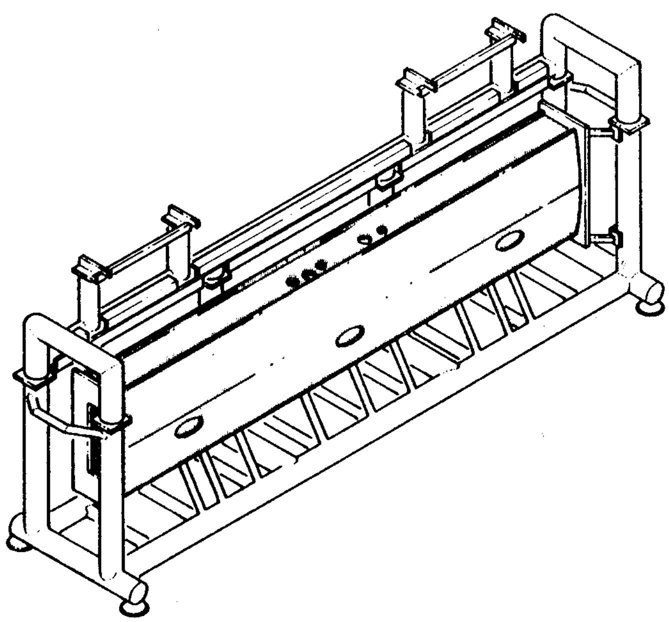


Figure 22

GROUND TEST PLAN

The current plan for structural verification of the composite center wing is to conduct comprehensive tests on a full-scale center wing box. These tests will use an existing test fixture (provided from an earlier NASA contract) suitably modified to improve load introduction capability so that representative loading will be achieved throughout the box span. The planned tests include verification of:

- o Stiffness (EI and GJ)
- o Static Strength (multiple conditions)
- o Durability Compliance (flight-by-flight spectra)
- o Damage Tolerance (delaminations/impact)
- o Repairability (static and cyclic)
- o Residual Strength (limit load)
- o Failure (growth potential)

At the completion of these tests a teardown inspection of the structure will be conducted. A final report will correlate test results with analytical predictions.

Limit Load	• **Upbending**
	• **Downbending**
Ultimate Load	• **Front Spar Shear**
	• **Rear Spar Shear**
Durability	• **Two Lifetimes**
	• **Flight By Flight**
Damage Tolerance	• **Damage & Repairs**
Residual Strength	• **Static**
	• **Multiple Sites**
Failure	• **Static**
	• **Single Site**

Figure 23

PROGRAM BENEFITS SUMMARY

The conclusive demonstration of technology readiness for use of advanced composites in primary wing structure of a large transport aircraft clears a major hurdle for production incorporation. Effective solutions to such concerns as major load transfer joints, fuel containment, lightning protection, and damage tolerance are expected. Development and demonstration of low-cost, reliable tooling to fabricate complex, highly-loaded structure, and test verification of the resulting wing box are vital. Cost and weight tracking provide a solid basis for calculating potential savings on future aircraft.

In summary, the successful completion of this program will provide industry management with the confidence needed to make production commitments for use of composites in large transport wing structure.

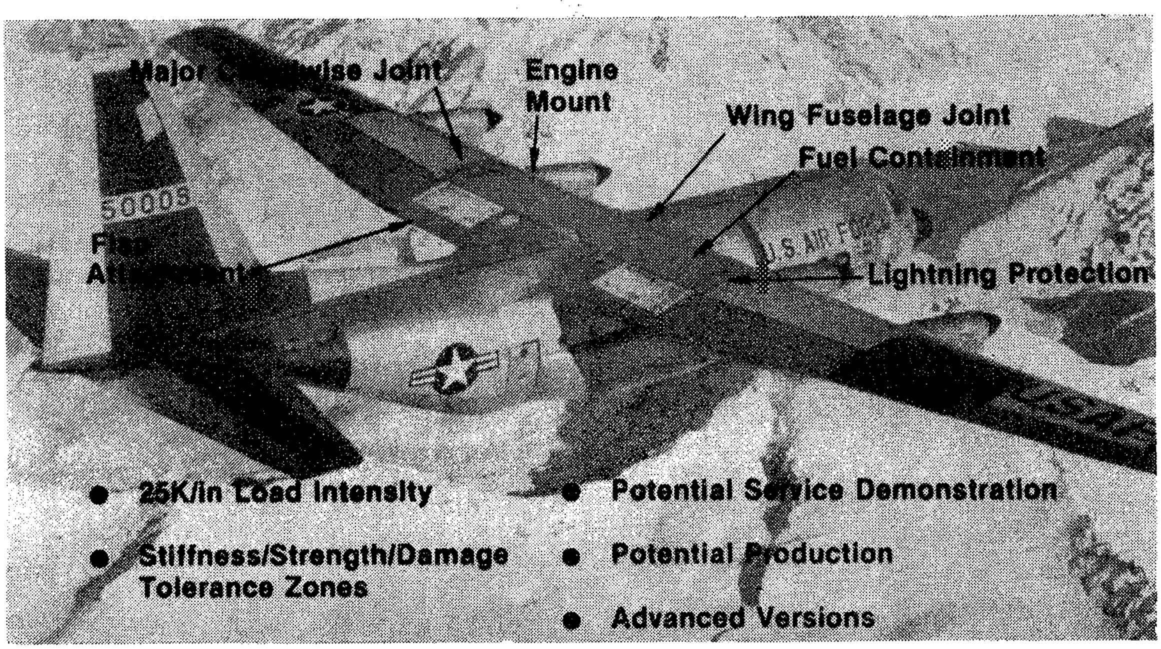


Figure 24

1. Report No. NASA CR-172360	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle ACEE COMPOSITE STRUCTURES TECHNOLOGY Papers by Lockheed-California Company and Lockheed-Georgia Company		5. Report Date August 1984
		6. Performing Organization Code
7. Author(s) Compiled by A. M. James		8. Performing Organization Report No.
		10. Work Unit No.
9. Performing Organization Name and Address Lockheed-Corporation Lockheed-California Company P. O. Box 551 Burbank, California 91520		11. Contract or Grant No. NAS1-17698
		13. Type of Report and Period Covered Contractor Report
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546		14. Sponsoring Agency Code 534-06-13

15. Supplementary Notes

Langley technical monitor: M. B. Dow

16. Abstract

The NASA Aircraft Energy Efficiency (ACEE) Composite Primary Aircraft Structures Program has made significant progress in the development of technology for advanced composites in commercial aircraft. Under NASA sponsorship, commercial airframe manufacturers have demonstrated technology readiness and cost effectiveness of advanced composites for secondary and medium primary components and have initiated a concerted program to develop the data base required for efficient application of safety-of-flight wing and fuselage structure. The third special oral review of the ACEE Composites Programs was held in Seattle, Washington, on August 13-16, 1984. The conference included comprehensive reviews of all composites technology development programs by ACEE Composites contractors. In addition, special sessions included review of selected NASA-sponsored research, and several important Department of Defense programs in composite materials and structures. The papers were compiled in five documents. Papers prepared by personnel from Boeing Commercial Airplane Company, Douglas Aircraft Company, and both Lockheed-California Company and Lockheed-Georgia Company are contained in NASA CR-172358, CR-172359, and CR-172360, respectively. Papers on selected NASA-sponsored research are contained in NASA CP-2321. Papers on selected Department of Defense programs in composites are contained in NASA CP-2322.

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